NACA

RESEARCH MEMORANDUM

ANALYSIS OF FACTORS AFFECTING SELECTION AND DESIGN OF AIR-COOLED SINGLE-STAGE TURBINES FOR TURBOJET ENGINES

III - ENGINE DESIGN-POINT PERFORMANCE

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON
September 16, 1954

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SUMMARY

Turbine blade cooling permits engine design freedom which can be exploited to improve both the turbine and the over-all turbojet-engine performance. Therefore, an analysis was made to determine the thrust and thrust specific fuel consumption for both afterburning and nonafterburning turbojet engines equipped with single-stage air-cooled turbines operating at a flight Mach number of 2 and an altitude of 50,000 feet. Wide ranges of coolant-flow ratios and turbine design specifications (turbine-inlet temperature, tip speed, and hub-tip radius ratio) were considered. In the interest of exploring outer limits as well as minimizing cooling requirements, the turbines were designed for high specific work commensurate with the assigned efficiency through the use of high turbine aerodynamic limits and high turbine-exit whirl. Other components were correspondingly considered to operate with high aerodynamic limits commensurate with the assigned efficiencies.

The results of this analysis showed that the thrust of turbojet engines may vary in excess of 300 percent of the minimum value obtained for the nonafterburning engines and 100 percent of the minimum value obtained for the afterburning engines as the turbine design specifications are varied over the range considered. At turbine-inlet temperatures of approximately 2000° R, little or no performance improvements are possible by increasing the turbine tip speed and, therefore, the compressor pressure ratio. However, if the turbine-inlet temperature and the turbine blade tip speed are increased, large improvements in thrust can be obtained because of simultaneous increases in tail-pipe nozzle jet velocity and weight flow. The higher turbine blade tip speed increases the turbine blade stresses such that at a tip speed of 1700 feet per second, root stresses for a practical blade metal taper exceed 50,000 pounds per square inch. However, maximum thrust increases in excess of 80 and 100 percent for the uncooled nonafterburning and afterburning engines, respectively, are possible by changing the turbine blade tip speed from 1100 to 1700 feet per second. In all cases, the losses

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due to cooling are small compared with the potential gains. At high turbine-inlet temperatures, the effect of coolant flow on the thrust specific fuel consumption of the nonafterburning engine is negligible.

At high turbine-inlet temperatures, the thrust per unit of turbine frontal area can also be increased, while the thrust specific fuel consumption is decreased at constant turbine blade stress, if the turbine blade tip speed and hub-tip radius ratio are increased simultaneously. This characteristic, resulting from the increasing compressor pressure
ratio which improves the turbine weight-flow capacity and thrust specific 3360
fuel consumption, suggests the use of high turbine hub-tip radius ratios at high turbine-inlet temperatures.

For the single-stage turbine operating at high turbine-inlet temperatures, turbine hub-tip radius ratios above approximately 0.70 for the nonafterburning and above 0.75 for the afterburning engines are desirable if the engine frontal area is a criterion of merit, since the engine weight-flow limitations imposed by the combustor and afterburner are relieved as a result of the higher compressor pressure ratios.

INTRODUCTION

In order to better evaluate the type and direction of research leading to future engine developments, exploratory analyses are required to determine the performance potentials of turbojet engines. Therefore, an analytical study was made to investigate the performance of engines equipped with single-stage turbines operating over wide ranges of turbine-inlet temperatures, tip speeds, and hub-tip radius ratios. The investigation was conducted at a flight Mach number of 2, and somewhat futuristic component aerodynamic limits were chosen commensurate with the assigned efficiencies. Since the freedom of considering a wide range of turbine design variables is afforded by use of turbine blade cooling to control the effective blade-metal strength, the analysis is of particular value in guiding turbine-cooling research.

Several analyses of the performance of both cooled and uncooled turbojet engines operating over wide ranges of engine design specifications are available (ref. 1). Adequate procedures were not available for analyzing and limiting each engine component at the time these analyses were made. Therefore, the engine performance was necessarily evaluated by using independent engine specifications which frequently led to designs for which it was impossible to pass the specified weight flow, the required component Mach numbers differed from those values which were commensurate with the component efficiencies, the cooling requirements were inaccurate, or a combination of these conditions existed. Thus, the performance results of many of the engines considered are not directly comparable.

Because of recent component research and the development of a simplified procedure for evaluating turbine work (ref. 2), it is relatively easy to relate engine performance to turbine-inlet temperature, turbine tip speed, and hub-tip radius ratio for specified component efficiencies and aerodynamic limits. In addition, cooling requirements can, in turn, be related to these variables. For these reasons, a series of investigations has been undertaken by the NACA to determine the performance of turbojet engines with single-stage air-cooled turbines for a wide range of turbine-inlet temperatures, tip speeds, and hub-tip radius ratios. The investigation involves a range of cooling-air flows (considered to be sufficient for most coolant-passage configurations which might be proposed) such that a variety of coolant-passage configurations may be incorporated. In addition, high turbine aerodynamic limits and turbine-exit whirl were utilized in the interest of obtaining the highest turbine specific work commensurate with the efficiency assigned, in order to explore the outer limits for single-stage turbine engines employing turbine blade cooling. The application of high turbine aerodynamic limits and turbine-exit whirl will also minimize cooling requirements. The engines investigated were made comparable by utilizing the same aerodynamic limits and efficiencies throughout. A general and complete discussion of the investigation is given in reference 1.

The analytical procedures for this investigation are presented in reference 3. The computed compressor pressure ratios and engine and turbine weight-flow capacities are presented in reference 1. The compressor pressure ratios and weight flows given in reference 1 are used along with the analytical techniques of reference 3 to compute the over-all engine performance for design-point operation at a flight Mach number of 2.0 at 50,000 feet altitude. The purpose of the present report is to present this over-all engine performance expressed in terms of the engine thrust and thrust specific fuel consumption. The following ranges in the turbine design specifications are considered: turbine-inlet temperature, 2000° to 3500° R; actual turbine blade tip speed, 1100 to 1700 feet per second; turbine hub-tip radius ratio, 0.55 to 0.75; and coolant-flow ratio, zero to 0.15.

PROCEDURES

The analytical techniques employed to evaluate the performance of turbojet engines equipped with single-stage air-cooled turbines are presented in detail in reference 3. The engines analyzed consist of an inlet diffuser, a compressor, a combustor, a turbine, and a nozzle. For the case of thrust augmentation, the engines have, in addition, a diffuser after the turbine, and an afterburner. (Schematic diagrams of unaugmented and augmented engines are shown in fig. 1.) The assumptions and constants used in the analysis are discussed in references 1 and 3. They are briefly presented in appendix A.

Each design point is independent of the actual engine size and has been specified by the turbine design specifications, that is, the turbine-inlet temperature, tip speed, and hub-tip radius ratio. In addition, in the interest of providing results suitable for incorporating a variety of coolant-passage configurations, a range of coolant-flow ratios (ratio of turbine coolant to compressor weight flow) was used for each combination of the turbine design specifications. The coolant-flow range was considered to be sufficient for most coolant-passage configurations proposed. It was assumed that the cooling air was bled at an intermediate stage of the compressor such that the total work done on 3360
the cooling air was equivalent to the work required to compress it to the compressor-outlet conditions. This differs from compressor-exit bleed in that pumping work is done on the cooling air as it flows within the turbine.

The values of the turbine aerodynamic limits and the turbine-exit whirl were chosen in order to obtain high turbine specific work (see appendix B for the definition of terms and symbols) commensurate with the assigned efficiency, and to minimize cooling requirements. Since the turbine design specifications and the coolant-flow ratio were chosen as the independent variables, the analysis was initiated by specifying a combination of these variables. From these independent variables, the turbine specific work was first computed. This, in turn, provided sufficient information for computing the compressor pressure ratio. An aerothermodynamic analysis was then employed to determine the engine performance from the compressor pressure ratio and the turbine-inlet temperature, and from the specified values for the aerodynamic limits and efficiencies of each component. The aerodynamic limits and efficiencies were required so that the weight-flow capacity or required area and losses for each component could be computed simultaneously. In order to assure that all engines analyzed were comparable, the same aerodynamic limits were used throughout. These limits were chosen commensurate with the assigned efficiencies.

RESULTS AND DISCUSSION

Variations of Thrust per Unit of Turbine Frontal Area and Thrust Specific Fuel Consumption with Turbine Design Specifications and Coolant-Flow Ratios

It is convenient to present first the engine performance in terms of the total thrust per unit of turbine frontal area as based upon the turbine weight-flow capacity and the thrust specific fuel consumption. Finally, the thrust is corrected or qualified according to the weight-flow capacity of the various components. The engine performance in terms of thrust per unit of turbine frontal area and thrust specific fuel consumption is given in table I for the complete range of turbine design specifications and turbine coolant-flow ratios. The corresponding values of the turbine weight flows, pressure ratios for each component, ratios of combustor to turbine frontal area, ratios of afterburner to turbine frontal area, engine temperature ratios, and fuel-air ratios are listed in table I of reference 1.

<u>Separate effects of each turbine design specification and coolant-</u>
3360 <u>flow ratio.</u> - Typical variations in the performance of the nonafterburning engine with each of the turbine design specifications and coolant-flow ratios are illustrated in figure 2. The performance variations are presented in terms of the percentage increases in both the thrust per unit of turbine frontal area and the thrust specific fuel consumption over the values for a turbine-inlet temperature of 2000° R, tip speed of 1300 feet per second, hub-tip radius ratio of 0.70, and zero coolant flow.

As shown by figure 2(a), the thrust improves with gains in both turbine-inlet temperature and turbine tip speed, but decreases with increases in the turbine hub-tip radius ratio. These changes in thrust are the results of changes in both the tail-pipe jet velocity and the turbine weight flow. For the conditions of figure 2, the compressor pressure ratio increases with the turbine-inlet temperature, turbine blade tip speed, and turbine hub-tip radius ratio (ref. 1). These increases in compressor pressure ratio result in rises in the pressure levels within the turbine and, therefore, higher turbine-exit mass velocity ρV, at least for constant turbine-inlet temperatures. This larger mass velocity entirely accounts for the increase in thrust with turbine blade tip speed. As the turbine hub-tip radius ratio becomes larger, however, the annular area decreases more than the mass velocity increases with a resulting reduction in the turbine weight flow. This decrease in turbine weight flow essentially accounts for the entire decrease in engine thrust per unit of turbine frontal area. For rising turbine-inlet temperatures, the turbine-exit temperature increases, tending to lower the mass velocity; however, higher turbine-inlet temperatures also tend to improve the pressure and mass velocity at the turbine exit because smaller turbine pressure drops occur, even though more work is required to drive the compressor for the higher compressor pressure ratios. The net effect of raising the turbine-inlet temperature for the cases of figure 2 is to increase the turbine weight flow. Thus the improvement in thrust with turbine-inlet temperatures shown on figure 2 is due to increases in the turbine weight flow as well as increases in the jet velocity. However, the increases in the turbine-limited specific weight flows with turbine-inlet temperatures (ref. 1) account for only approximately 15 percent of the thrust increase shown. It should be realized that figure 2 is only intended to be illustrative of trends and that the percentage increases in either turbine-limited specific weight flow or jet velocity will be somewhat different for other conditions.

The thrust specific fuel consumption (fig. 2(b)) increases with turbine-inlet temperature but decreases with increasing turbine blade tip speed and hub-tip radius ratio. In both cases, the decrease in the thrust specific fuel consumption is a direct result of the compressor-pressure-ratio increase. However, as the turbine-inlet temperature rises, the thrust specific fuel consumption increases chiefly because the increase in the jet velocity accompanying the higher temperature results in a larger increase in the jet kinetic energy (proportional to the square of the jet velocity) than in the jet momentum (proportional to the jet velocity). 3360

The effects of bleeding cooling air from the compressor are a decrease in thrust and an increase in thrust specific fuel consumption for the conditions of figure 2. This effect is, in general, a result of the reduction in the obtainable compressor pressure ratio for any combination of turbine design specifications without an increase in the turbine pressure ratio p_7'/p_4' (ref. 1). Thus, the tail-pipe pressure ratio decreases as the coolant flow becomes greater; the thrust is lowered and the thrust specific fuel consumption is increased. Only the highest coolant-flow ratio (0.15) is shown on figure 2, since the effect of coolant-flow ratio upon the engine performance varies essentially linearly with the coolant-flow ratio, as can be seen from table I.

Typical variations in the performance of the afterburning engines with each of the turbine design specifications and coolant-flow ratios are illustrated in figure 3 for an afterburner-outlet temperature of 3500° R. As in figure 2, the performance variations are presented in terms of the percentage increases in both the thrust per unit of turbine frontal area and the thrust specific fuel consumption over the values for a nonafterburning engine with a turbine-inlet temperature of 2000° R, tip speed of 1300 feet per second, hub-tip radius ratio of 0.70, and zero coolant flow.

The thrust per unit of turbine frontal area for the afterburning engine (fig. 3(a)) increases as the turbine-inlet temperature and tip speed become greater, but decreases as the turbine hub-tip radius ratio increases, as for the nonafterburning engine. However, two distinct differences between the afterburning and nonafterburning engine exist. First, the jet-nozzle temperature is unaffected by changes in the turbine-inlet temperature or turbine work (or compressor pressure ratio) because the afterburner temperature is specified at 3500° R. Second, an increase in turbine weight flow raises the percentage increase in the thrust per unit of frontal area more for the afterburning engine than for the nonafterburning engine because of the higher basic thrust of the afterburning engine. These differences are particularly noticeable as the turbine-inlet temperature or tip speed is changed.

As the turbine blade tip speed is increased, the thrust shown in figure 3 for the afterburning engine increases more than that shown in figure 2 for the nonafterburning engine. This is a reflection of the effect of the increase in the turbine-limited specific weight flow, as well as the increase in the jet velocity resulting from the higher compressor pressure ratios with a constant jet-nozzle temperature. In addition, with the rise in turbine-inlet temperature, the increase in the thrust of the afterburning engine is somewhat less than that for the nonafterburning engine, since for a constant afterburner-outlet temperature, the jet velocity increases only because of an increase in the jet-nozzle pressure ratio accompanying a rise in the compressor pressure ratio. For the afterburning engine, approximately one-half of the total thrust gain resulting from the higher turbine-inlet temperature is due to increases in the turbine-limited specific weight flow.

The thrust specific fuel consumption of the afterburning engine (fig. 3(b)) decreases with increase in the turbine blade tip speed and turbine hub-tip radius ratio, but somewhat less than for the nonafterburning engine. As the turbine-inlet temperature rises, the thrust specific fuel consumption decreases rather than increases as for the nonafterburning engine. This decrease in the thrust specific fuel consumption accompanies an increase in both the turbine pressure ratio p_7'/p_4' and the compressor pressure ratio p_3'/p_2' and, therefore, an increase in the jet-nozzle pressure ratio p_{11}'/p_∞ with a slight reduction in the total fuel-air ratio. The effect of coolant-flow ratio upon the afterburning engine is similar to that for the nonafterburning engine, although it is larger for the thrust specific fuel consumption and, in general, smaller for the thrust per unit of turbine frontal area.

Combined effects of turbine-inlet temperature, tip speed, and coolant-flow ratio at hub-tip radius ratio of 0.70. - The variation of the performance of both the afterburning and the nonafterburning engines over the complete range of turbine-inlet temperatures and tip speeds is shown in figure 4 for a hub-tip radius ratio of 0.70 and for coolant-flow ratios of zero and 0.15. In addition, in order to illustrate the potential performance gains over that of present design practice, a point is located on each the afterburning- and nonafterburning-engine maps at values of turbine design specifications near present design values for uncooled single-stage turbines.

It is interesting to note that, for a turbine-inlet temperature of 2000° R, little improvement in the engine performance is possible for the assigned flight Mach number of 2. However, by increasing the turbine-inlet temperature and the turbine blade tip speeds, large gains in thrust per unit of turbine frontal area can be obtained. For instance, for the uncooled nonafterburning engine, a rise in the turbine-inlet temperature from 2000° to 3500° R, accompanied by an increase in the turbine blade tip speed to 1700 feet per second, permits at least a 300-percent gain in thrust per unit of turbine frontal area. Similar changes in the turbine-inlet temperature and tip speed for the afterburning engine permit at least a 100-percent increase in the thrust per unit of turbine frontal area.

It is also shown in figure 4 that the thrust per unit of turbine frontal area increases most rapidly with turbine-inlet temperature for the highest turbine blade tip speeds, and with turbine blade tip speed 3360
for the highest turbine-inlet temperature. In fact, for a turbine-inlet temperature of 3500° R, a change in the turbine blade tip speed from 1100 to 1700 feet per second will permit increases in the thrust per unit of turbine frontal area of approximately 80 and 100 percent for the nonafterburning and afterburning engines, respectively. Thus, the results in figure 4 illustrate that an incentive exists for simultaneously increasing both the turbine-inlet temperature and the tip speed.

For the afterburning engine, the thrust specific fuel consumption, in general, decreases significantly as either the turbine-inlet temperature or the turbine blade tip speed increases. However, for the nonafterburning engine, as the turbine-inlet temperature rises, at constant turbine blade tip speed, the thrust specific fuel consumption increases. In order to maintain constant thrust specific fuel consumption as the turbine-inlet temperature is elevated, the turbine blade tip speed must also be increased. For example, as the turbine-inlet temperature and blade tip speed are increased from 2000° R and 1300 feet per second to 2500° R and 1500 feet per second, respectively, the thrust specific fuel consumption remains essentially unchanged for the uncooled engine. As the turbine-inlet temperature is increased to 3000° or 3500° R, the turbine blade tip speed would necessarily exceed 1700 feet per second to maintain the thrust specific fuel consumption.

As either the turbine-inlet temperature or the turbine blade tip speed is increased, the cooling requirements, in general, become greater. As illustrated by figure 4, the effect of cooling air is to shift the performance toward lower thrust and higher thrust specific fuel consumption. However, it is important to notice that the effect of cooling air on the thrust specific fuel consumption of the nonafterburning engine diminishes as the turbine-inlet temperature rises. In fact, at a turbine-inlet temperature of 3500° R and at tip speeds from 1100 to 1700 feet per second, the increase in thrust specific fuel consumption for the nonafterburning engine is only approximately 1 percent for an increase in coolant flow from zero to 0.15, while the reduction in thrust per unit of turbine frontal area is between approximately 15 and 20 percent. However, for the afterburning engine with a turbine-inlet temperature of 3500° R, the thrust specific fuel consumption increases by 7 percent as the coolant-flow ratio changes from zero to 0.15, while the thrust per unit of turbine frontal area is reduced by about 5 and 13 percent for turbine blade tip speeds of 1100 and 1700 feet per second, respectively. For a coolant-flow ratio of 0.15, which is probably large even for the most severe conditions of figure 4, the net performance gains possible by increasing the turbine-inlet temperature and tip speed above present design values are large. Even for the nonafterburning engine, it should be possible to maintain the thrust specific fuel consumption at present design value while obtaining significant increases in thrust by cooling the turbine and employing turbine-inlet temperatures somewhat higher than 2000° R.

Effect of compressor pressure ratio. - In order to illustrate the variations in the compressor pressure ratio as well as the effects of these variations on the engine performance, the results of figure 4 for zero coolant flow are presented again in figure 5. Figure 5 is identical to the corresponding part for zero coolant flow in figure 4, except that cross plots of the compressor pressure ratio are included. As shown in figure 5, as either the turbine-inlet temperature or the turbine blade tip speed increases, the compressor pressure ratio increases. However, the increase in compressor pressure ratio is considerably greater with increasing turbine blade tip speed. As the compressor pressure ratio increases, for constant turbine-inlet temperature, the thrust specific fuel consumption, in general, decreases. However, at a turbine-inlet temperature of 2000° R, a minimum thrust specific fuel consumption is obtained as the turbine blade tip speed and, therefore, the compressor pressure ratio, is increased to about 7 for the nonafterburning engine.

The performance map formed by the compressor-pressure-ratio and the turbine-inlet-temperature lines in figure 5 is independent of the number of turbine stages and, therefore, represents a generalization of the results for any turbine which is designed for the exit critical gas velocity ratio and exit whirl energy employed herein. For a multistage turbine, either the turbine blade tip speed, or the inlet relative Mach number, or both, must be lower than the corresponding values for the single-stage turbine employed herein. Therefore, the turbine blade tip speed given in figure 5 can be considered as superimposed for a single-stage turbine. Similar lines of turbine blade tip speeds could be superimposed for any particular multistage turbine design. Since the pressure-ratio lines are general, the only effects of utilizing multistage turbines is to extend the range of compressor pressure ratios obtainable and, therefore, the thrust and thrust specific fuel consumptions obtainable, or to reduce the required turbine blade tip speeds and turbine blade stresses for the range of performance on figure 5 (necessarily at the expense of larger turbine blade heat-transfer surface area and probably turbine weight). As shown on figure 5, for a hub-tip radius ratio of 0.70, at present-day temperatures of approximately 2000° R and for a flight Mach number of 2.0 or higher, no performance gains are possible by extending the range of compressor pressure ratios.

Effect of maintaining constant turbine specific weight flow. - The performance results when the turbine specific weight flow is kept constant at 18.5 pounds per second per square foot (the limiting value at a turbine-inlet temperature of 2000° R, turbine hub-tip radius ratio of 0.10, and a turbine blade tip speed of 1100 ft/sec) over the entire range of turbine-inlet temperature and turbine tip speeds are shown in figure 6. The map applies to conditions of zero coolant flow and a hub-tip radius ratio of 0.70. Because the turbine specific weight flow is kept constant, the aerodynamic limits in each component, in general, must differ for each engine design. Inasmuch as the results on figure 5 are for the same conditions, except that the turbine specific weight flow is always at the limiting value, which varies for each point on the map, the effect of the turbine-limited specific weight flow on the performance can be determined by comparing the results of figures 5 and 6. Since the results are reduced to a common weight flow in figure 6, the thrust per unit of turbine frontal area is proportional to the specific thrust (thrust/lb of air flow). The range of obtainable thrusts in figure 5 is considerably greater than that in figure 6 because of the permissible turbine-limited specific-weight-flow variations.

Increases in the turbine-limited specific weight flow account primarily for the large thrust gain of the afterburning engine as the turbine-inlet temperature or turbine blade tip speed is increased. In fact, the maximum thrust shown in figure 5 for the afterburning engine exceeds that in figure 6 for the afterburning engine by about 80 percent, as a result of the increase in the turbine-limited specific weight flow. The largest gains in the turbine-limited specific weight flow and thrust per unit of turbine frontal area due to increased turbine blade tip speed (or compressor pressure ratio) are obtained at the highest turbine-inlet temperature. Figure 6 illustrates that as the turbine-inlet temperature of the nonafterburning engine rises, large gains in thrust are obtained because of the increase in nozzle-jet velocity. However, a comparison of figures 5 and 6 shows that at higher turbine blade tip speeds, a large part of the thrust gains accompanying rising turbine-inlet temperatures is also caused by turbine-limited specific-weight-flow increases. The effect of increases in the turbine-limited specific weight flow is to improve the maximum thrust shown in figure 6 for the nonafterburning engine by about 80 percent. The same increase was obtained for the afterburning engine.

At low turbine-inlet temperatures, little or no improvement in the turbine-limited specific weight flow is obtained as the turbine blade tip speed is increased. Thus, for both the afterburning and nonafterburning engines, indications are that the incentive for increasing both the turbine-inlet temperature and the turbine blade tip speed is caused not only by improvements in specific thrust, as would be expected but also by the increase in the turbine-limited specific weight flow.

Combined effect of turbine tip speed and hub-tip radius ratio. - As the turbine hub-tip radius ratio is changed from 0.70, which is representative of the value used in present-day design practice, both the turbine annular area and the obtainable turbine work change. If the turbine hub-tip radius ratio decreases, the flow area increases and the specific-weight-flow capacity of the turbine tends to become greater. However, for constant turbine blade tip speed, this increase in flow area is accompanied by higher turbine blade stresses and lower blade root speeds. The decrease in the blade root speed reduces the obtainable turbine specific work and compressor pressure ratio for an aerodynamically limited turbine.

In order to illustrate these changes, figure 7 presents the thrust per unit of turbine frontal area and the thrust specific fuel consumption of the uncooled engines for two turbine-inlet temperatures (2500° and 3500° R) and a range of turbine blade tip speeds and hub-tip radius ratios. Since cooling air does not affect the general trends illustrated in figure 7, the performance is presented only for the uncooled engines, even though cooling air would necessarily be required. The effect of coolant-flow ratio upon the performance results of figure 7 is to shift the results to lower thrust per unit of turbine frontal area and higher thrust specific fuel consumption, as in figure 4. Lines of constant compressor pressure ratio are also cross-plotted on the performance maps to illustrate the changes in compressor pressure ratio. As before, the compressor pressure-ratio lines generalize the performance such that it is independent of the number of turbine stages employed. The lines of constant compressor pressure ratio almost coincide with the lines of constant thrust specific fuel consumption, the slight difference being caused by changes in the pressure drop within the various engine components that arise from the weight-flow changes.

For a turbine-inlet temperature of 3500° R (fig. 7(b)), as the turbine hub-tip radius ratio decreases, the thrust per unit of turbine frontal area increases to a maximum value at a hub-tip radius ratio of approximately 0.60. The improvement in thrust resulting from a reduction in the turbine hub-tip radius ratio from 0.75 to the value corresponding to the maximum thrust varies between approximately 10 and 16 percent for the nonafterburning engine and approximately 6 and 10 percent for the afterburning engine. The change in thrust accompanying a decrease in turbine hub-tip radius ratio is a reflection of the changes in the turbine-limited specific weight flow (ref. 1). The thrust per unit of turbine frontal area increases with decreasing hub-tip radius ratio until a maximum is reached, and thereafter decreases because both the specific thrust and the turbine-exit mass velocity are reduced.

Although similar trends must exist for a turbine-inlet temperature of 2500° R (fig. 7(a)), this peak thrust per unit of turbine frontal area was not reached for the range of hub-tip radius ratios considered. As the turbine hub-tip radius ratio decreases from 0.75 to 0.55 the percentage thrust increase at a turbine-inlet temperature of 2500° R is substantially larger than at 3500° R. In fact, for 2500° R, the maximum thrust changes obtained in figure 7(a) with changes in the turbine hub-tip radius ratio vary from approximately 20 to 30 percent for the afterburning engines and approximately 25 to 50 percent for the nonafterburning engines (the largest changes occur for turbine blade tip speeds of 1700 ft/sec). In all cases, the thrust specific fuel consumption increases with decreases in the turbine hub-tip radius ratio because of the reduction in the compressor pressure ratio.

Turbine Blade Stresses at Various Turbine Performance Levels

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The turbine blade centrifugal stresses are indicative of the severity of the turbine blade mechanical design and govern cooling requirements for a given turbine-inlet temperature. In addition, the stress limitations impose a limit upon the range of turbine design specifications. Figure 8 illustrates the variation in stresses with turbine design specifications for two turbine-inlet temperatures (2500° and 3500° R) and zero coolant-flow ratio. Since the turbine blade stresses are dependent upon both the turbine tip speed and hub-tip radius ratio, the lines of constant blade root stress are superimposed upon the performance maps constructed with these turbine design specifications as parameters. The stresses were computed on the basis of a linear metal-area taper from blade hub to tip, with the tip metal area 40 percent of that at the hub. This amount of taper was chosen because it can be obtained with present fabrication procedures for convection-air-cooled-blade configurations.

The constant-stress lines shown in figure 8 illustrate the necessity of designing for high turbine blade stresses if the high-performance engine designs are to be obtained with a single-stage turbine. For instance, at a turbine blade tip speed of 1700 feet per second and a hub-tip radius ratio of 0.70, the tapered blade root stress is slightly above 55,000 pounds per square inch. Reducing the stresses to 30,000 pounds per square inch or below (as in present design practice) greatly penalizes the performance. As would be expected from the results already obtained, considerably larger thrust improvements are possible by increasing the turbine blade stresses at the higher than at the lower turbine-inlet temperatures.

The slopes of the constant-stress lines shown in figure 8 are considerably different for the two turbine-inlet temperatures presented. For the turbine-inlet temperature of 2500° R (fig. 8(a)), the nonafterburning-engine thrust per unit of turbine frontal area can be increased along the lines of constant stress, but at the expense of greater thrust specific fuel consumption. However, for a turbine-inlet temperature of 3500° R (fig. 8(b)), the constant-stress lines possess a more desirable slope, which corresponds to an increasing thrust per unit of turbine frontal area accompanied by a reduction in thrust specific fuel consumption. This difference is caused by the larger increases in the turbine-limited specific weight flows obtained by increasing the turbine blade tip speed at higher than at lower turbine-inlet temperatures.

For the afterburning engines (figs. 8(a) and (b)), the slopes of the constant-stress lines are, in general, similar in direction but, again, considerably more favorable for the higher turbine-inlet temperature. Points of maximum thrust are obtained on some of the stress lines
3360 shown on figure 8, particularly for the afterburning engine with high turbine blade-root stresses. In all cases, maximum thrust points would be obtained if the range of variables were extended. It is important to point out that, if the thrust per unit of turbine frontal area becomes larger and thrust specific fuel consumption decreases along a constant-stress line, the turbine blade tip speed and hub-tip radius ratio increase. The thrust gains resulting in this manner are primarily caused by the more rapid increase in the turbine-limited specific weight flow with turbine blade tip speed than its decrease with turbine hub-tip radius ratio. The specific-fuel-consumption reduction results from the corresponding increases in compressor pressure ratio. Therefore, the constant-blade-root-stress lines of figure 8(b) indicate that for a turbine-inlet temperature of 3500° R it is desirable to design the turbine with short blades (high hub-tip radius ratio) and high tip speeds, rather than to use long blades and low tip speeds. This indication is limited, of course, at a maximum turbine hub-tip radius ratio corresponding to the peak thrust point on the stress lines as occurs only for the high stresses in figure 8(b). At a turbine-inlet temperature of 2500° R (fig. 8(a)), the lines of constant turbine blade-root stress give less information concerning the selection of a combination of turbine blade tip speed and hub-tip radius ratio. This is particularly true with the nonafterburning engines, since the stress lines are inclined such that the advantages of thrust per unit of turbine frontal area increases must be weighed against the disadvantages of the thrust specific fuel consumption increases. In all cases, however, a more rigorous study of the cooling requirements and the problems associated with the turbine blade stresses in the design of turbines is required in order to draw positive conclusions.

Typical Performance Variations for Constant Blade-Root Stress

The performance of the nonafterburning engine for three turbine-inlet temperatures (2500°, 3000°, and 3500° R) is presented in figure 9 for a constant blade root stress of 40,000 pounds per square inch and zero coolant flow. The 2500° and 3500° R turbine-inlet temperature lines shown in figure 9 are extracted from the corresponding parts of figure 8. A comparison of these lines with the 3000° R turbine-inlet temperature line shows that the slope of the constant-stress lines for 3000° R is between those for 2500° and 3500° R. This exists for all values of the turbine blade root stresses.

In addition, figure 9 shows that the peak thrust points move continuously toward higher turbine hub-tip radius ratios as the turbine-inlet temperature is increased (for 3500° R, the peak is not obtained but would exist at some turbine hub-tip radius ratio greater than 0.75). Since this is accompanied by a rise in the turbine speed and, therefore, a rise in the compressor pressure ratio, the thrust specific fuel consumption decreases. Therefore, it is indicated that, as the turbine-inlet temperature is raised, the best design value of the turbine blade hub-tip radius ratio increases. In this manner, large thrust improvements are obtained with minor changes in the thrust specific fuel consumption. This characteristic exists because of the changes in the turbine-limited specific weight flow, which are completely different at the low and high turbine-inlet temperatures.

Variations of Thrust Per Unit of Engine Frontal Area and Thrust Specific Fuel Consumption with Turbine Design Specifications

Influence of engine component sizes on engine performance and selection of turbine design specifications. - In the foregoing discussion, the engine thrust was presented in terms of the thrust per unit of turbine frontal area as based upon the turbine-limited specific weight flow. It was shown that the turbine-limited specific weight flow has an important effect upon the thrust per unit of turbine frontal area. Therefore, since the thrust per unit of engine frontal area may be a criterion of merit, it is important that the weight-flow limitations imposed by the other engine components be considered. Since aerodynamic limits were imposed upon the engine components of the present analysis, either the maximum specific weight flow of each component or the ratio of required frontal area for each component to the turbine frontal area was computed.

The engine weight-flow limitations imposed by all components except the tail-pipe nozzle are presented in reference 1, with the assumption that the weight flow is limited by the component or components having the largest frontal area. Increases in the frontal areas of the components due to structural or accessory requirements are not accounted for in reference 1. However, once these requirements are specified, they may be included in determining engine weight-flow limitations from the data presented in table I of reference 1. In general, for complete expansion, the tail pipe limits the engine specific weight flow. However, separate calculations have been made showing that, if incomplete expansion is employed such that the tail-pipe nozzle area is no larger than the turbine frontal area, the thrust is esssentially unchanged from that for complete expansion.

In reference 1 it is shown that the compressor limits the specific weight flow of the uncooled engines only at the highest values of the turbine-inlet temperatures and tip speeds studied (turbine-inlet temperature and turbine blade tip speed above approximately 3500° R and 1700 ft/sec, respectively). In addition, this limitation exists only for turbine hub-tip radius ratios below 0.70. However, at these conditions, cooling air will certainly be required, and therefore the uncooled engine is somewhat unrealistic. Since cooling-air flow reduced the turbine-limited specific weight flow, the cooled engine is probably unlimited by the compressor for all conditions studied.

For an inlet diffuser ram recovery of 0.85, the inlet diffuser specific weight flow is less than that for the uncooled turbine only at the highest turbine-inlet temperature and tip speed. Since for these conditions the inlet diffuser specific weight flow is greater than that for the compressor, it imposes no weight-flow limit. However, as shown in reference 1, the combustor and afterburner impose weight-flow limitations for a large number of combinations of the turbine design specifications. In fact, the afterburner limits the engine specific weight flow for all conditions studied except for moderate and high turbine tip speeds at a 2000° R turbine-inlet temperature, as shown in figure 7(a) of reference 1. The combustor specific-weight-flow capacity exceeds that of the turbine for turbine hub-tip radius ratios above approximately 0.70 for low turbine tip speeds. For higher tip speeds, the limiting turbine hub-tip radius ratio is below 0.70, the exact value varying considerably with temperature.

The effect of these weight-flow limitations imposed by the combustor and the afterburner on the thrust is illustrated in figure 10, where the thrust per unit of frontal area is plotted against the turbine hub-tip radius ratio for both the afterburning and the nonafterburning engines operating at a turbine-inlet temperature of 2000° R and a turbine blade tip speed of 1300 feet per second. The dashed lines in figure 10 represent the thrust per unit of turbine frontal area as employed in the previous figures. The solid line is the thrust per unit of engine frontal area. For the nonafterburning engine, the thrust per unit of engine frontal area is the thrust per unit of turbine frontal area divided by the ratio of the combustor to turbine frontal area (table I of ref. 1) for the cases in which the combustor to turbine frontal area is greater than 1. Similarly, for the afterburning engine, the portion of the curve of the thrust per unit of engine frontal area to the left of the intersection point is the thrust per unit of turbine frontal area divided by the ratio of the afterburner to turbine frontal area (table I of ref. 1).

For both the afterburning and the nonafterburning engines, the thrust per unit of engine frontal area decreases as the hub-tip radius ratio is reduced from the value at the intersection point because the engine specific weight flow decreases (variations in the engine specific weight flow are similar to those shown in fig. 8 of ref. 1). In addition, the thrust specific fuel consumption increases as the hub-tip radius ratio decreases, because of the drop in the compressor pressure ratio. Thus, if the thrust per unit of engine frontal area and the thrust specific fuel consumption are of primary importance, turbine blade hub-tip radius ratios less than the value at the intersection point are undesirable. In addition, it is apparent that the thrust per unit of engine frontal area for any combination of turbine-inlet temperature and tip speed is a maximum at this intersection point, where the turbine-limited specific weight flow and the combustor or afterburner specific-weight-flow capacity are equal.

The effects of increasing the combustor and afterburner reference velocities are illustrated in reference 1. In the case of the nonafterburning engine, the use of higher combustor velocities provides little or no improvement in the engine specific-weight-flow capacity because of the increasing combustor pressure drop. However, large gains in the afterburning-engine specific weight flows are possible if the afterburner velocities are increased. The effects of these gains on the engine performance were not determined herein, because the afterburner velocities required are considerably in excess of the current design values of approximately 400 feet per second. However, limited improvements may be realized in the near future by increasing the afterburner velocity above 500 feet per second assumed herein.

Maximum obtainable thrust per unit of engine frontal area. - The maximum obtainable thrust per unit of engine frontal area as evaluated for the turbine hub-tip radius ratio where the turbine-limited specific weight flow and the combustor (in the case of the nonafterburning engine) or the afterburner (in the case of the afterburning engine) specific-weight-flow capacity are equal is presented in figure 11 for all combinations of turbine-inlet temperatures and tip speeds at zero coolant flow. For those cases in which the turbine-limited specific weight flow exceeds the afterburner specific-weight-flow capacity, the maximum thrust for the afterburning engine was determined by extrapolating the computed results to higher turbine hub-tip radius ratios. The sharp break or limit in the nonafterburning-engine thrust per unit of frontal area at a turbine-inlet temperature of 3500° R and a high turbine blade tip speed is caused by the compressor specific-weight-flow limit. It is not intended that the conditions of figure 11 be presented as the most desirable for design, even though the thrust per unit of engine frontal area may be an important criterion, since the thrust specific fuel consumption can be further reduced by allowing the thrust to decrease. In addition, no attention is given to the required turbine blade stresses.

As previously noted, for constant turbine blade root stress, there is an incentive to increase the turbine hub-tip radius ratios and turbine blade tip speeds as the turbine-inlet temperature rises in order to obtain simultaneously a larger thrust and a smaller thrust specific fuel consumption. For the nonafterburning engine, which has peak values of the thrust per unit of engine frontal area at turbine hub-tip radius ratios of 0.70 or below, it may be desirable to employ higher turbine hub-tip radius ratios in favor of better thrust specific fuel consumption, or blade root stresses, or both. This, of course, reduces the thrust from the maximum obtainable values presented in figure 11. For the afterburning engines, the peak values of the thrust per unit of engine frontal area correspond to turbine hub-tip radius ratios of approximately 0.75 or above. The tendency toward higher turbine hub-tip radius ratios as previously discussed is, therefore, particularly desirable in this case.

Because of the higher turbine blade hub-tip radius ratios, the compressor pressure ratios for the afterburning engines of figure 11 are greater than those for the nonafterburning engines, while the corresponding stresses of the afterburning engines are less. The effect of this difference in compressor pressure ratio is to improve the thrust specific fuel consumption of the afterburning engine. A principal result shown by figure 11 is that the maximum obtainable thrust per unit of engine frontal area for the afterburning engine relative to that of the nonafterburning engine is lower than the corresponding values of the thrust per unit of turbine-frontal area as shown in figure 4 for a turbine hub-tip radius ratio of 0.70. This illustrates that the weight-flow limitations of the afterburning engine are more severe than for the nonafterburning engine.

It is impossible to consider all factors which may impose limits on the design-point performance of either afterburning or nonafterburning engines. For any specific application, the effects of factors other than those studied herein must certainly be considered. Important among these factors is the operation flexibility required for each engine component if the engine is to operate satisfactorily at off-design conditions. The flexibility required may, in some cases, dictate the need for employing conservative aerodynamic limits for one or more of the components at the design point. This, in turn, may affect the design-point performance obtainable.

CONCLUDING REMARKS

The results presented herein show that the performance of afterburning and nonafterburning engines equipped with single-stage turbines and operating at a flight Mach number of 2 at an altitude of 50,000 feet varies widely with the turbine design specifications. Emphasis is particularly placed upon the effect of the weight-flow capacity of the various components on the thrust per unit of frontal area and the conditions which lead to favorable changes in both the thrust and the thrust specific fuel consumption. The general trends indicated that the conclusions can be summarized as follows:

1. As the turbine design specifications are varied over the range considered, the thrust of turbojet engines may vary in excess of 300 percent of the minimum value obtained for the nonafterburning engines and 100 percent of the minimum value obtained for the afterburning engines.

2. At turbine-inlet temperatures around 2000° R, very little improvement in the performance of either afterburning or nonafterburning engines is possible by increasing the turbine tip speed and, therefore, the compressor pressure ratio. However, as the turbine-inlet temperature and turbine tip speed are increased, large improvements in thrust can be obtained because of simultaneous increases in tail-pipe nozzle 3360
jet velocity and weight flow. For instance, for a turbine hub-tip radius ratio of 0.70 and a turbine-inlet temperature of 3500° R, the thrust per unit of turbine frontal area can be increased by about 80 and 100 percent for the nonafterburning and afterburning engines, respectively, by increasing the turbine blade tip speed from 1100 to 1700 feet per second. This increase in turbine blade tip speed is accompanied by an increase in the turbine blade root stress to approximately 55,000 pounds per square inch. In all cases, the losses due to cooling are small compared with the potential gains, and at high turbine-inlet temperatures, the effect of coolant flow on the thrust specific fuel consumption of the nonafterburning engine is negligible.

3. As the turbine hub-tip radius ratio is reduced at constant turbine blade tip speed, the thrust per unit of turbine frontal area increases because of the gain in the turbine-limited specific weight flow, but at the expense of greater thrust specific fuel consumption. The largest percentage increases in thrust obtained in this way occurred at the lowest turbine-inlet temperatures. At a turbine-inlet temperature of 2500° R, the thrust per unit of turbine frontal area can be increased from approximately 20 to 50 percent by changing the turbine hub-tip radius ratio from 0.75 to 0.55.

4. At the highest value of the turbine-inlet temperature studied (3500° R), the thrust per unit of turbine frontal area can be improved while the thrust specific fuel consumption is lowered without increasing the blade centrifugal stresses by simultaneously increasing the turbine hub-tip radius ratio and the turbine blade tip speed. This illustrates the incentive for designing with high turbine hub-tip radius ratios at the high turbine-inlet temperatures. In general, for the nonafterburning engine at the lower turbine-inlet temperatures, however, the thrust can be increased at constant blade root stresses only by raising the thrust specific fuel consumption.

5. The afterburner imposes a more severe limit on the engine specific-weight-flow capacity and, therefore, the maximum obtainable thrust per unit engine frontal area, than does the combustor. This adversely affects the afterburning engine in comparison with the nonafterburning engine. For both the afterburning and nonafterburning engines, it is undesirable to design for turbine hub-tip radius ratios for which either the afterburner or combustor impose limits on the engine specific

weight flow if the engine frontal area is an important criterion. For the afterburning engine, this suggests particularly high turbine hub-tip radius ratios (0.75 and above), and for the nonafterburning engine above approximately 0.70.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, June 25, 1954

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APPENDIX A

ASSUMPTIONS AND CONSTANTS

The following assumptions apply to the turbine analysis herein:

(1) Simplified radial equilibrium

(2) Free-vortex blading 3360

(3) Use of density at mean radial position for computing weight flow

(4) Stagnation temperature invariant with radial position at turbine entrance

(5) Hub, mean, and tip radii at turbine exit equal to respective values at turbine entrance

(6) Constant value of 4/3 for ratio of specific heats

(7) Adiabatic flow through turbine rotor

(8) Work done by turbine on cooling air between compressor inlet and tips of turbine blades equivalent to work required to compress cooling air to compressor-outlet conditions

(9) Mechanical friction and work required by accessories negligible

(10) Pressure head associated with tangential component of turbine absolute efflux velocity dissipated by struts that secure tail cone, and resulting loss charged to turbine adiabatic efficiency

(11) No effect of cooling-air efflux from turbine blade tips on turbine work or adiabatic efficiency

(12) Ratio of turbine blade metal area at tip to that at root equal to 0.40

(13) Turbine-exit whirl corresponding to value of -0.41 for ratio of whirl velocity to wheel speed at radial position where product of whirl energy loss per pound and total weight flow expresses total whirl loss

(14) Turbine blade metal density equal to 0.30 pound per cubic inch

In calculating the maximum specific weight mass flow of the compressor, the following assumptions were imposed:

(15) Use of simplified radial equilibrium in analysis of flow at inlet to first compressor rotor

(16) Computation of weight flow through compressor inlet at mean radial position

(17) Use of solid-body-rotation guide vanes in compressor-inlet analysis

The following additional assumptions were imposed on the pertinent components:

(18) Constant value of 4/3 for ratio of specific heats in combustor analysis

(19) Variation of combustor pressure ratio with inlet Mach number and temperature ratio according to experimental loss characteristics of typical low-loss turbojet combustor

(20) Entrance of combustion gases and cooling air into zone downstream of turbine in which mixing occurred at same static pressure

(21) Constant-flow area in mixing zone for combustion gases and cooling air

(22) Constant-area combustion section in afterburner

(23) Friction and flame-holder losses in afterburner equal to inlet dynamic head

(24) For calculating thrust, jet nozzle was assumed to provide complete expansion to ambient pressure

The following values of constants were employed herein:

Compressor-inlet hub-tip radius ratio	0.4
Compressor adiabatic efficiency	0.83
Compressor rotor-inlet relative Mach number at blade tip . . .	1.1
Compressor-inlet axial critical velocity ratio at hub	0.7
Ratio of combustor inner radius to turbine tip radius	0.25
Combustion efficiency	0.95
Combustor reference velocity limit, ft/sec	150
Turbine adiabatic efficiency	0.83
Turbine-inlet relative Mach number at hub	0.8
Turbine-exit axial critical velocity ratio	0.7
Afterburner combustion efficiency	0.88

Afterburner-inlet velocity, ft/sec	500
Afterburner-exit stagnation temperature, $^{\circ}R$	3500
Afterburner diffuser efficiency	0.85
Afterburner friction and flame-holder drag coefficient	1.0
Tail-pipe nozzle efficiency	0.95

APPENDIX B

DEFINITIONS AND SYMBOLS

The following terms, which were employed in the text in the interest of brevity, are defined below:

3360 Compressor specific weight flow	Corrected weight flow per unit compressor frontal area that an aerodynamically limited compressor can pass
Engine specific weight flow	Corrected weight flow of compressor divided by area of largest engine component among the following: inlet diffuser, compressor, combustor, turbine, and afterburner (where applicable)
Inlet diffuser specific weight flow	Corrected weight flow of diffuser divided by diffuser frontal area
Turbine design specifications	Turbine-inlet temperature, blade tip speed, and blade hub-tip radius ratio
Turbine specific work	Turbine work per pound of working fluid
Turbine-limited specific weight flow	Corrected weight flow through compressor as limited by turbine, divided by turbine frontal area

The following symbols are used in this report:

A	frontal area of any engine component, sq ft
C	ratio of cooling air to compressor weight flow
F	thrust, lb
p'	stagnation pressure, lb/sq ft
r	radius, ft
T'	stagnation temperature, $^{\circ}R$
U	blade speed, ft/sec
V	velocity, ft/sec

w weight flow, lb/sec

δ ratio of stagnation to standard pressure, $p'/2116$

σ blade root stress, lb/sq in.

ρ density, lb/cu ft

θ ratio of stagnation to standard temperature, $T'/518.4$ 3360

Subscripts:

C compressor

E engine

h hub

T turbine

1 to 11 engine stations as shown in figure 1

∞ free stream

REFERENCES

1. Rossbach, Richard J., Schramm, Wilson B., and Hubbartt, James E.: Analysis of Factors Affecting Selection and Design of Air-Cooled Single-Stage Turbines for Turbojet Engines. I - Turbine Performance and Engine Weight-Flow Capacity. NACA RM E54C22, 1954.

2. Cavicchi, Richard H., and English, Robert E.: A Rapid Method for Use in Design of Turbines within Specified Aerodynamic Limits. NACA TN 2905, 1953.

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TABLE I. - TABULATION OF RESULTS

(a) Nonafterburning engine.

Coolant-flow ratio, C	Turbine hub-tip radius ratio, r_h/r_t	Turbine tip speed, U_t, 1100 ft/sec		Turbine tip speed, U_t, 1300 ft/sec		Turbine tip speed, U_t, 1500 ft/sec		Turbine tip speed, U_t, 1700 ft/sec	
		Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T
Turbine-inlet temperature, T_4', 2000° R									
0	0.55	1.625	559	1.546	605	1.496	626	1.464	614
	.60	1.594	531	1.519	566	1.472	573	1.437	547
	.65	1.566	492	1.497	517	1.458	512	1.415	478
	.70	1.543	444	1.474	466	1.427	451	1.407	392
	.75	1.505	400	1.427	410	1.406	375	1.416	299
0.05	0.55	1.655	531	1.572	574	1.527	589	1.501	573
	.60	1.625	502	1.543	537	1.508	535	1.488	509
	.65	1.603	465	1.520	488	1.485	482	1.470	444
	.70	1.568	420	1.496	440	1.464	423	1.468	362
	.75	1.527	380	1.470	385	1.461	349	1.487	278
0.10	0.55	1.678	509	1.618	541	1.566	550	1.552	532
	.60	1.654	479	1.586	502	1.558	501	1.536	473
	.65	1.622	441	1.570	456	1.528	450	1.526	410
	.70	1.604	396	1.535	411	1.517	390	1.540	333
	.75	1.562	360	1.514	359	1.514	323	1.565	254
0.15	0.55	1.704	483	1.651	509	1.622	509	1.606	490
	.60	1.682	450	1.629	470	1.597	463	1.595	437
	.65	1.668	415	1.620	424	1.579	417	1.595	375
	.70	1.641	375	1.581	385	1.576	362	1.629	302
	.75	1.600	340	1.549	338	1.578	296	1.671	229
Turbine-inlet temperature, T_4', 2500° R									
0	0.55	1.729	780	1.648	903	1.574	1010	1.511	1093
	.60	1.696	757	1.610	876	1.540	967	1.480	1032
	.65	1.665	721	1.578	826	1.511	905	1.444	960
	.70	1.637	667	1.546	767	1.471	843	1.417	870
	.75	1.589	623	1.507	704	1.439	754	1.395	738
0.05	0.55	1.735	752	1.662	856	1.592	951	1.534	1022
	.60	1.707	726	1.631	827	1.558	908	1.508	964
	.65	1.675	690	1.595	781	1.532	848	1.466	897
	.70	1.656	633	1.567	727	1.492	793	1.439	809
	.75	1.607	589	1.529	663	1.462	704	1.420	687
0.10	0.55	1.750	722	1.679	813	1.614	897	1.559	952
	.60	1.726	688	1.653	777	1.581	854	1.531	894
	.65	1.695	654	1.620	734	1.556	792	1.495	836
	.70	1.674	598	1.585	684	1.517	740	1.465	751
	.75	1.623	561	1.551	625	1.489	656	1.454	635
0.15	0.55	1.779	681	1.700	767	1.639	839	1.589	887
	.60	1.738	658	1.674	735	1.610	799	1.563	826
	.65	1.718	617	1.645	688	1.583	743	1.520	779
	.70	1.690	570	1.606	646	1.544	689	1.500	691
	.75	1.643	532	1.572	586	1.515	608	1.489	583

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TABLE I. - Continued. TABULATION OF RESULTS

(a) Concluded. Nonafterburning engine.

Coolant-flow ratio, C	Turbine hub-tip radius ratio, r_h/r_t	Turbine tip speed, U_t, 1100 ft/sec		Turbine tip speed, U_t, 1300 ft/sec		Turbine tip speed, U_t, 1500 ft/sec		Turbine tip speed, U_t, 1700 ft/sec	
		Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T
Turbine-inlet temperature, T_4', 3000° R									
0	0.55	1.854	970	1.764	1163	1.682	1357	1.607	1563
	.60	1.820	948	1.723	1151	1.640	1347	1.570	1528
	.65	1.786	917	1.691	1108	1.603	1299	1.540	1458
	.70	1.758	860	1.653	1046	1.568	1234	1.502	1378
	.75	1.710	817	1.608	988	1.532	1142	1.483	1226
0.05	0.55	1.864	923	1.773	1106	1.691	1288	1.616	1463
	.60	1.831	907	1.729	1091	1.654	1265	1.583	1426
	.65	1.795	876	1.699	1047	1.615	1214	1.550	1362
	.70	1.760	822	1.664	992	1.574	1160	1.519	1284
	.75	1.716	781	1.623	931	1.540	1071	1.492	1145
0.10	0.55	1.869	890	1.781	1048	1.704	1213	1.634	1364
	.60	1.836	869	1.734	1037	1.670	1182	1.599	1325
	.65	1.800	833	1.709	987	1.629	1140	1.562	1271
	.70	1.768	785	1.679	935	1.587	1084	1.534	1194
	.75	1.726	744	1.628	876	1.557	996	1.507	1063
0.15	0.55	1.882	844	1.795	990	1.718	1136	1.656	1266
	.60	1.845	824	1.763	966	1.680	1110	1.620	1225
	.65	1.818	787	1.724	929	1.654	1054	1.583	1178
	.70	1.782	746	1.685	886	1.610	1007	1.549	1105
	.75	1.735	706	1.645	821	1.573	926	1.530	981
Turbine-inlet temperature, T_4', 3500° R									
0	0.55	1.992	1114	1.889	1389	1.796	1687	1.721	2012
	.60	1.947	1118	1.845	1388	1.755	1699	1.684	2019
	.65	1.908	1089	1.802	1369	1.719	1673	1.652	1968
	.70	1.875	1041	1.768	1319	1.682	1617	1.619	1893
	.75	1.827	999	1.727	1262	1.646	1528	1.592	1737
0.05	0.55	1.999	1068	1.888	1320	1.797	1589	1.726	1880
	.60	1.952	1067	1.851	1313	1.762	1594	1.691	1879
	.65	1.919	1032	1.808	1292	1.723	1564	1.657	1834
	.70	1.875	996	1.772	1247	1.688	1516	1.621	1772
	.75	1.825	955	1.725	1190	1.652	1426	1.597	1622
0.10	0.55	2.027	1007	1.897	1251	1.809	1493	1.733	1751
	.60	1.957	1011	1.857	1240	1.764	1496	1.696	1750
	.65	1.915	991	1.819	1214	1.731	1453	1.663	1703
	.70	1.873	956	1.774	1180	1.691	1423	1.628	1645
	.75	1.832	904	1.735	1116	1.653	1334	1.601	1511
0.15	0.55	2.000	980	1.900	1186	1.817	1398	1.745	1623
	.60	1.971	962	1.863	1169	1.777	1393	1.709	1619
	.65	1.920	941	1.828	1135	1.747	1347	1.669	1583
	.70	1.880	907	1.778	1110	1.698	1325	1.637	1524
	.75	1.829	865	1.734	1055	1.662	1242	1.605	1400

TABLE I. - Continued. TABULATION OF RESULTS

(b) Afterburning engine.

Coolant-flow ratio, C	Turbine hub-tip radius ratio, r_h/r_t	Turbine tip speed, U_t, 1100 ft/sec		Turbine tip speed, U_t, 1300 ft/sec		Turbine tip speed, U_t, 1500 ft/sec		Turbine tip speed, U_t, 1700 ft/sec	
		Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T	Thrust specific fuel consumption	Thrust per unit turbine frontal area, F/A_T
Turbine-inlet temperature, T_4', 2000° R									
0	0.55	2.398	1192	2.344	1347	2.321	1467	2.321	1537
	.60	2.374	1150	2.331	1286	2.317	1387	2.330	1430
	.65	2.356	1082	2.321	1204	2.319	1285	2.339	1315
	.70	2.342	994	2.307	1114	2.313	1176	2.371	1145
	.75	2.313	910	2.287	1006	2.322	1021	2.421	939
0.05	0.55	2.455	1177	2.395	1322	2.377	1425	2.383	1476
	.60	2.430	1128	2.380	1262	2.374	1339	2.396	1369
	.65	2.418	1059	2.376	1171	2.373	1244	2.406	1252
	.70	2.395	973	2.358	1085	2.370	1132	2.438	1087
	.75	2.361	893	2.343	974	2.388	974	2.498	887
0.10	0.55	2.504	1176	2.463	1293	2.440	1383	2.454	1416
	.60	2.490	1118	2.449	1225	2.443	1295	2.466	1311
	.65	2.468	1045	2.440	1138	2.437	1200	2.483	1190
	.70	2.458	951	2.420	1050	2.443	1079	2.522	1025
	.75	2.422	874	2.410	938	2.459	927	2.587	832
0.15	0.55	2.570	1162	2.530	1268	2.521	1331	2.530	1359
	.60	2.555	1097	2.517	1196	2.515	1245	2.543	1260
	.65	2.541	1026	2.515	1103	2.512	1155	2.569	1130
	.70	2.521	939	2.493	1020	2.517	1037	2.619	965
	.75	2.481	863	2.472	916	2.543	881	2.690	779
Turbine-inlet temperature, T_4', 2500° R									
0	0.55	2.260	1219	2.197	1446	2.142	1675	2.105	1891
	.60	2.236	1191	2.166	1425	2.120	1641	2.091	1837
	.65	2.209	1146	2.145	1366	2.105	1568	2.076	1760
	.70	2.188	1073	2.122	1289	2.081	1495	2.067	1646
	.75	2.147	1012	2.094	1202	2.065	1369	2.071	1447
0.05	0.55	2.303	1212	2.243	1415	2.190	1626	2.154	1819
	.60	2.281	1179	2.217	1388	2.167	1586	2.145	1758
	.65	2.253	1132	2.192	1329	2.153	1511	2.122	1682
	.70	2.237	1049	2.167	1259	2.126	1441	2.115	1564
	.75	2.192	988	2.139	1165	2.111	1310	2.117	1375
0.10	0.55	2.360	1202	2.296	1389	2.240	1586	2.210	1743
	.60	2.338	1157	2.273	1346	2.220	1537	2.198	1677
	.65	2.310	1110	2.247	1292	2.208	1452	2.177	1609
	.70	2.290	1026	2.217	1222	2.180	1384	2.167	1488
	.75	2.241	971	2.191	1130	2.162	1255	2.172	1301
0.15	0.55	2.431	1181	2.355	1360	2.307	1531	2.273	1676
	.60	2.393	1149	2.329	1322	2.285	1484	2.261	1602
	.65	2.372	1086	2.307	1254	2.265	1408	2.230	1536
	.70	2.346	1014	2.275	1192	2.238	1330	2.231	1408
	.75	2.298	954	2.244	1096	2.221	1198	2.236	1225

TABLE I. - Concluded. TABULATION OF RESULTS

(b) Concluded. Afterburning engine.

Coolant-flow ratio, C	Turbine hub-tip radius ratio, r_h/r_t	Turbine tip speed, U_t, 1100 ft/sec: Thrust specific fuel consumption	Turbine tip speed, U_t, 1100 ft/sec: Thrust per unit turbine frontal area, F/A_T	Turbine tip speed, U_t, 1300 ft/sec: Thrust specific fuel consumption	Turbine tip speed, U_t, 1300 ft/sec: Thrust per unit turbine frontal area, F/A_T	Turbine tip speed, U_t, 1500 ft/sec: Thrust specific fuel consumption	Turbine tip speed, U_t, 1500 ft/sec: Thrust per unit turbine frontal area, F/A_T	Turbine tip speed, U_t, 1700 ft/sec: Thrust specific fuel consumption	Turbine tip speed, U_t, 1700 ft/sec: Thrust per unit turbine frontal area, F/A_T
Turbine-inlet temperature, T_4', 3000° R									
0	0.55	2.179	1207	2.101	1486	2.038	1782	1.987	2120
	.60	2.149	1190	2.069	1486	2.008	1798	1.965	2116
	.65	2.119	1162	2.043	1451	1.983	1763	1.950	2063
	.70	2.097	1101	2.016	1386	1.959	1706	1.928	1998
	.75	2.052	1055	1.978	1329	1.935	1609	1.922	1829
0.05	0.55	2.227	1186	2.147	1453	2.082	1737	2.027	2037
	.60	2.196	1175	2.110	1451	2.054	1734	2.007	2025
	.65	2.164	1144	2.086	1410	2.028	1691	1.989	1974
	.70	2.138	1082	2.059	1351	1.995	1645	1.970	1906
	.75	2.092	1039	2.024	1285	1.972	1545	1.957	1742
0.10	0.55	2.275	1181	2.195	1423	2.130	1687	2.079	1955
	.60	2.243	1163	2.162	1416	2.104	1670	2.056	1933
	.65	2.211	1123	2.136	1369	2.074	1634	2.032	1892
	.70	2.184	1066	2.109	1312	2.043	1578	2.015	1814
	.75	2.140	1020	2.066	1244	2.020	1476	2.000	1657
0.15	0.55	2.341	1159	2.256	1389	2.189	1629	2.139	1869
	.60	2.305	1138	2.227	1367	2.158	1615	2.113	1843
	.65	2.277	1097	2.194	1329	2.137	1557	2.087	1805
	.70	2.243	1048	2.158	1280	2.101	1511	2.063	1727
	.75	2.192	999	2.122	1202	2.071	1410	2.052	1569
Turbine-inlet temperature, T_4', 3500° R									
0	0.55	2.113	1168	2.029	1489	1.957	1853	1.906	2271
	.60	2.076	1181	1.993	1504	1.929	1892	1.886	2322
	.65	2.043	1159	1.961	1500	1.904	1891	1.869	2305
	.70	2.016	1119	1.936	1462	1.880	1857	1.851	2259
	.75	1.974	1082	1.903	1415	1.857	1782	1.840	2118
0.05	0.55	2.164	1153	2.070	1455	1.998	1792	1.947	2176
	.60	2.123	1159	2.040	1461	1.974	1821	1.924	2215
	.65	2.094	1131	2.007	1453	1.943	1813	1.904	2200
	.70	2.060	1100	1.976	1421	1.918	1784	1.883	2163
	.75	2.012	1064	1.938	1368	1.894	1703	1.871	2023
0.10	0.55	2.216	1135	2.123	1421	2.050	1734	1.991	2085
	.60	2.178	1131	2.090	1421	2.014	1759	1.965	2118
	.65	2.138	1120	2.058	1405	1.993	1729	1.944	2097
	.70	2.100	1088	2.021	1380	1.957	1721	1.922	2059
	.75	2.061	1037	1.986	1319	1.931	1634	1.907	1927
0.15	0.55	2.279	1121	2.182	1386	2.107	1672	2.050	1986
	.60	2.251	1109	2.147	1380	2.072	1687	2.020	2013
	.65	2.201	1093	2.116	1354	2.050	1651	1.992	2001
	.70	2.160	1063	2.070	1337	2.007	1646	1.968	1958
	.75	2.108	1022	2.031	1283	1.977	1565	1.948	1829

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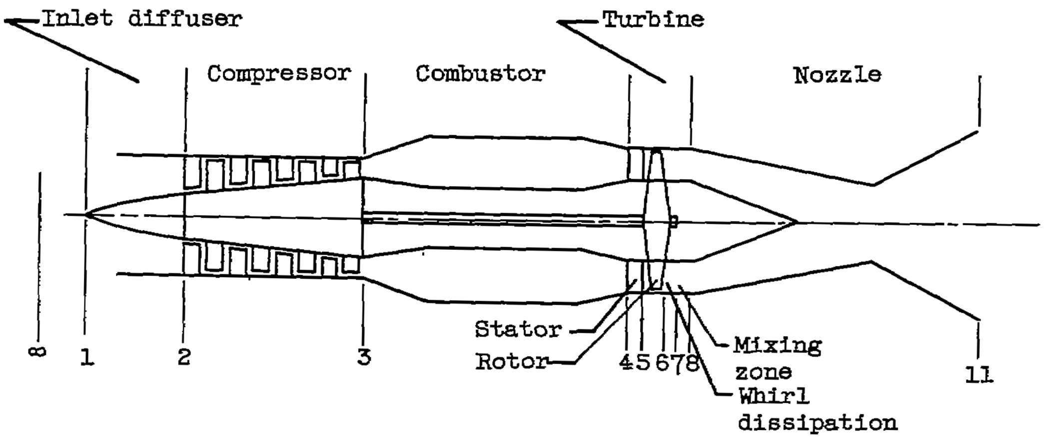
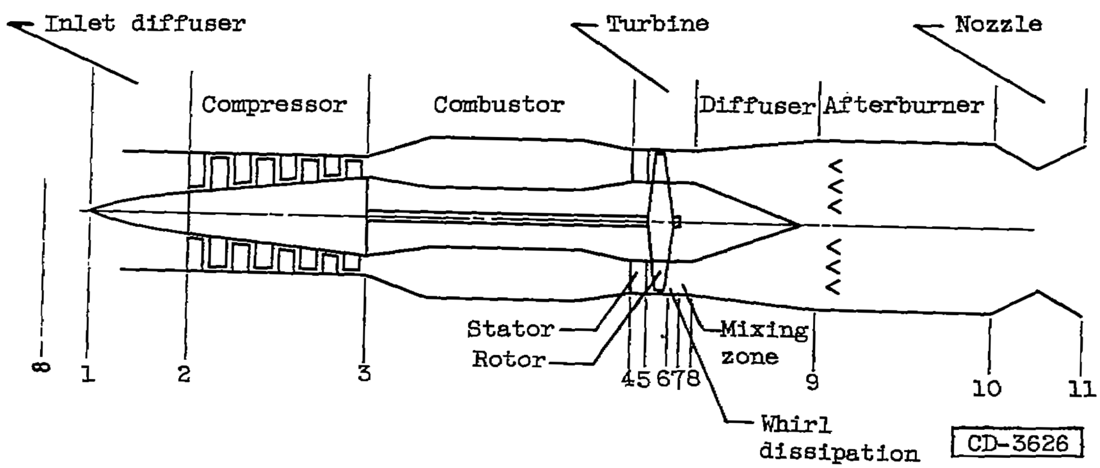


(a) Nonafterburning engine.



(b) Afterburning engine.

Figure 1. - Schematic diagram of two turbojet engines.

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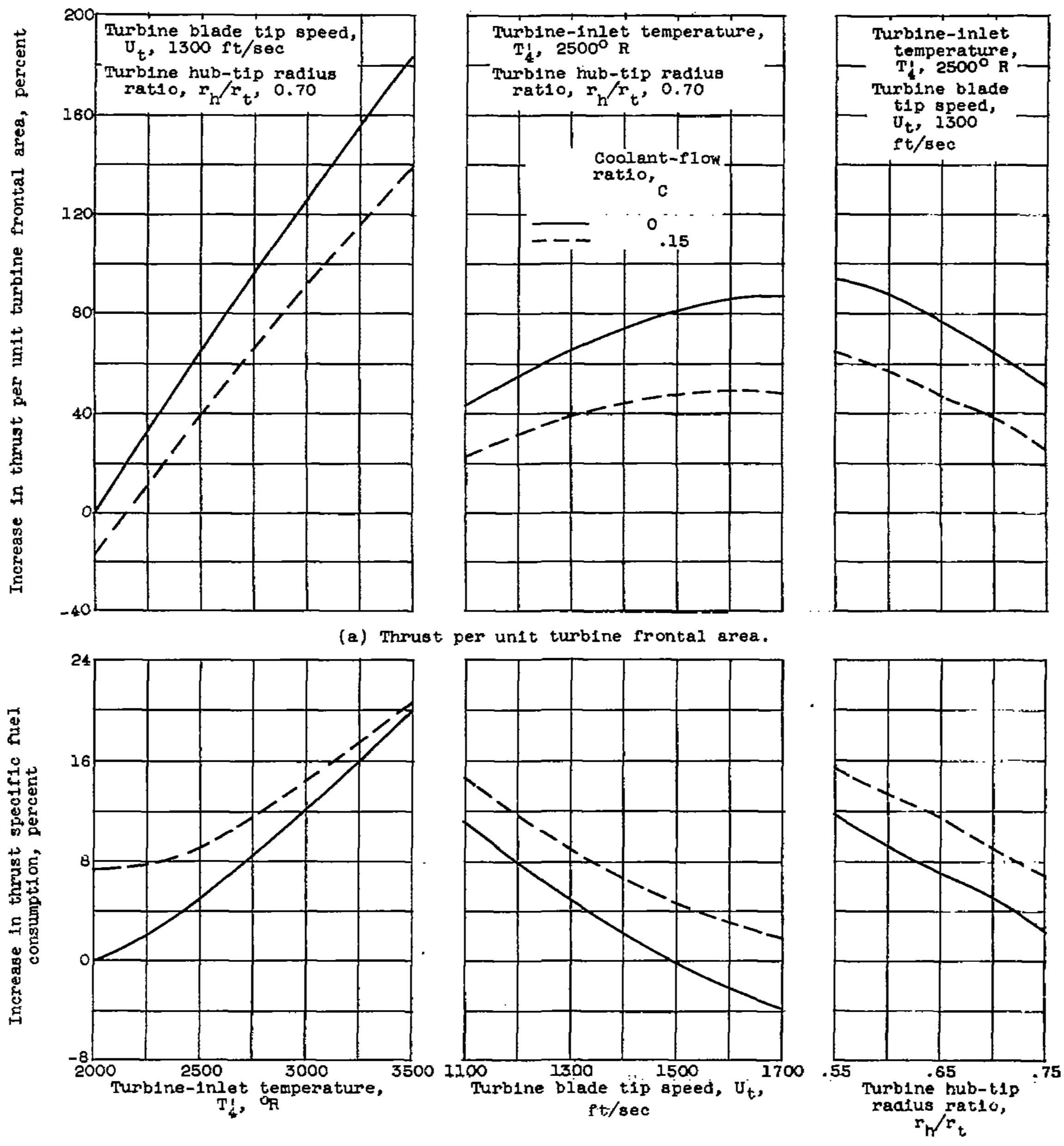

(a) Thrust per unit turbine frontal area.

(b) Thrust specific fuel consumption.

Figure 2. - Typical variations in nonafterburning-engine performance with turbine design variables.

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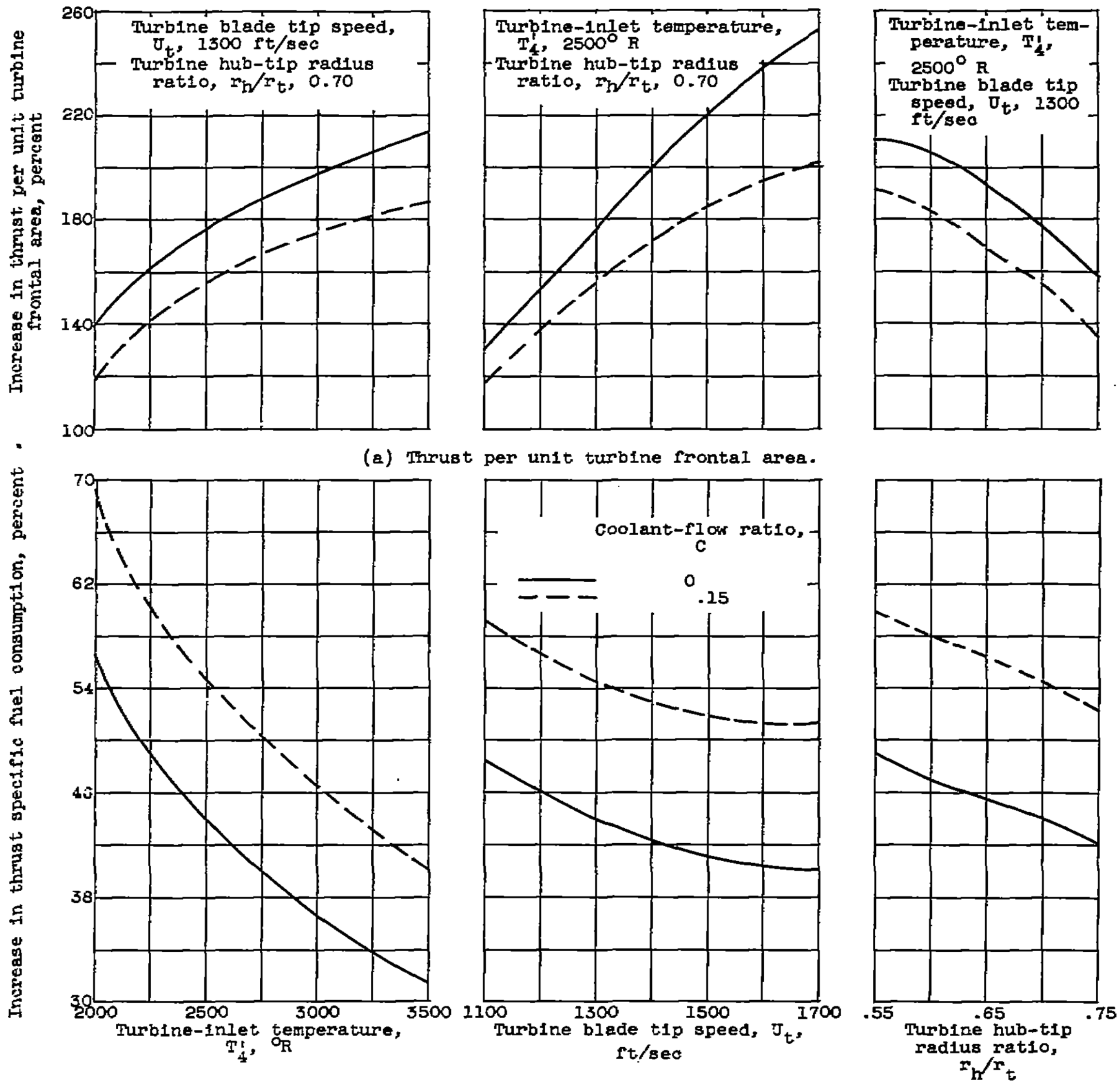

(a) Thrust per unit turbine frontal area.

(b) Thrust specific fuel consumption.

Figure 3. - Variations in afterburning-engine performance with turbine design variables. Afterburner-outlet temperature T_{10}', 3500° R.

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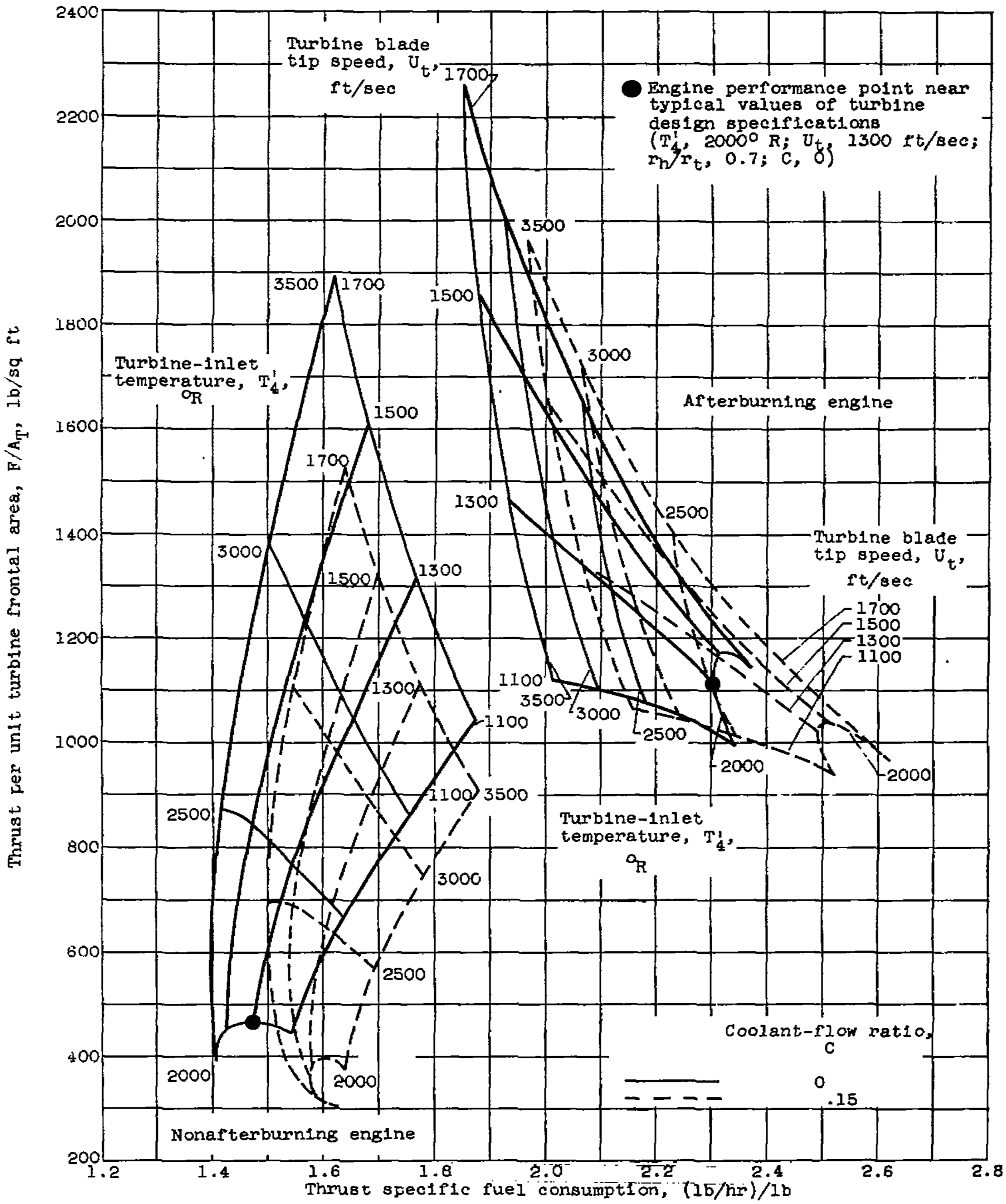

Figure 4. - Afterburning- and nonafterburning-engine performance for range of turbine design variables. Turbine hub-tip radius ratio r_h/r_t, 0.70; afterburner-outlet temperature T_{10}', 3500° R.

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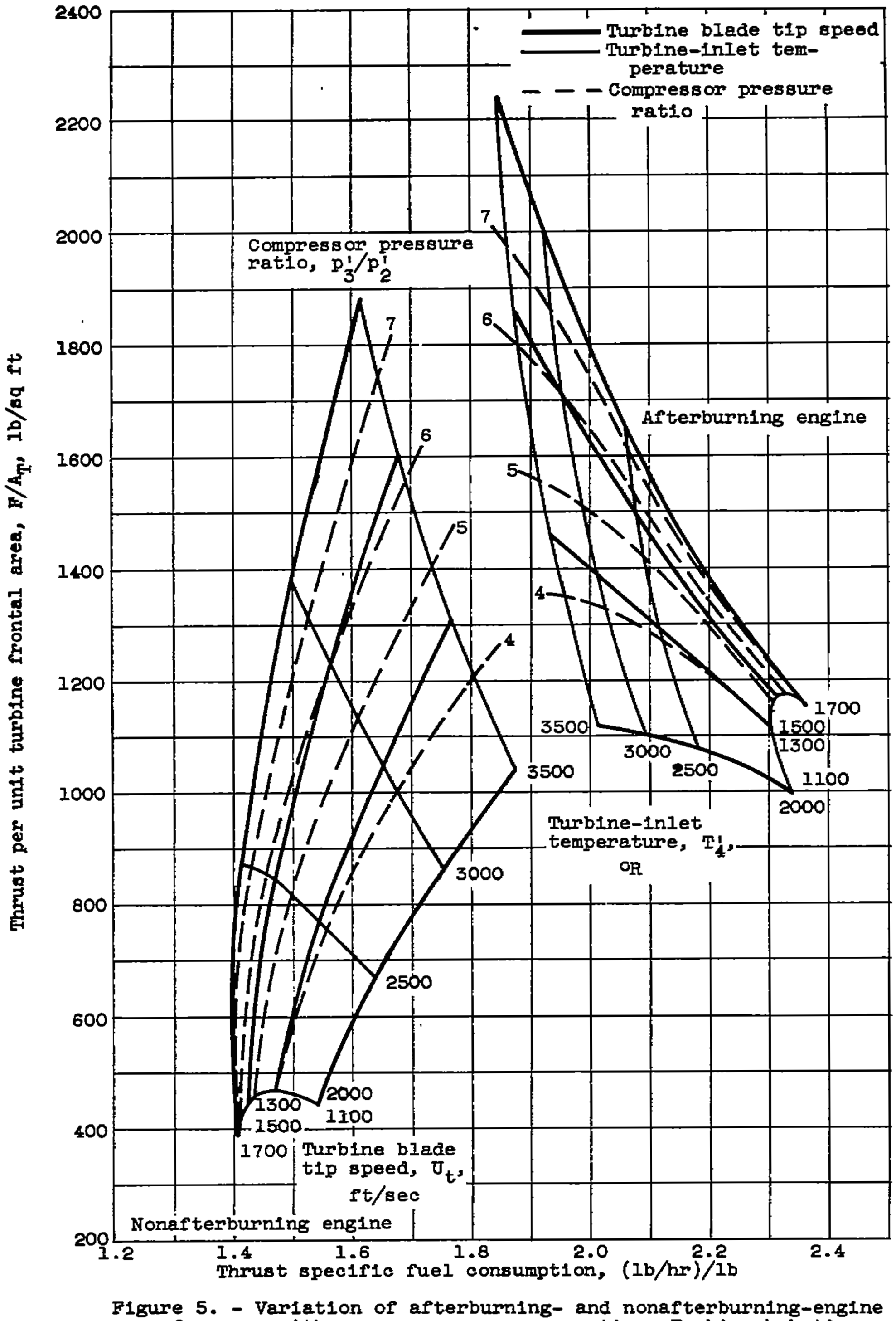

Figure 5. - Variation of afterburning- and nonafterburning-engine performance with compressor pressure ratio. Turbine hub-tip radius ratio r_h/r_t, 0.70; coolant-flow ratio C, 0; afterburner-outlet temperature T_{10}', 3500° R.

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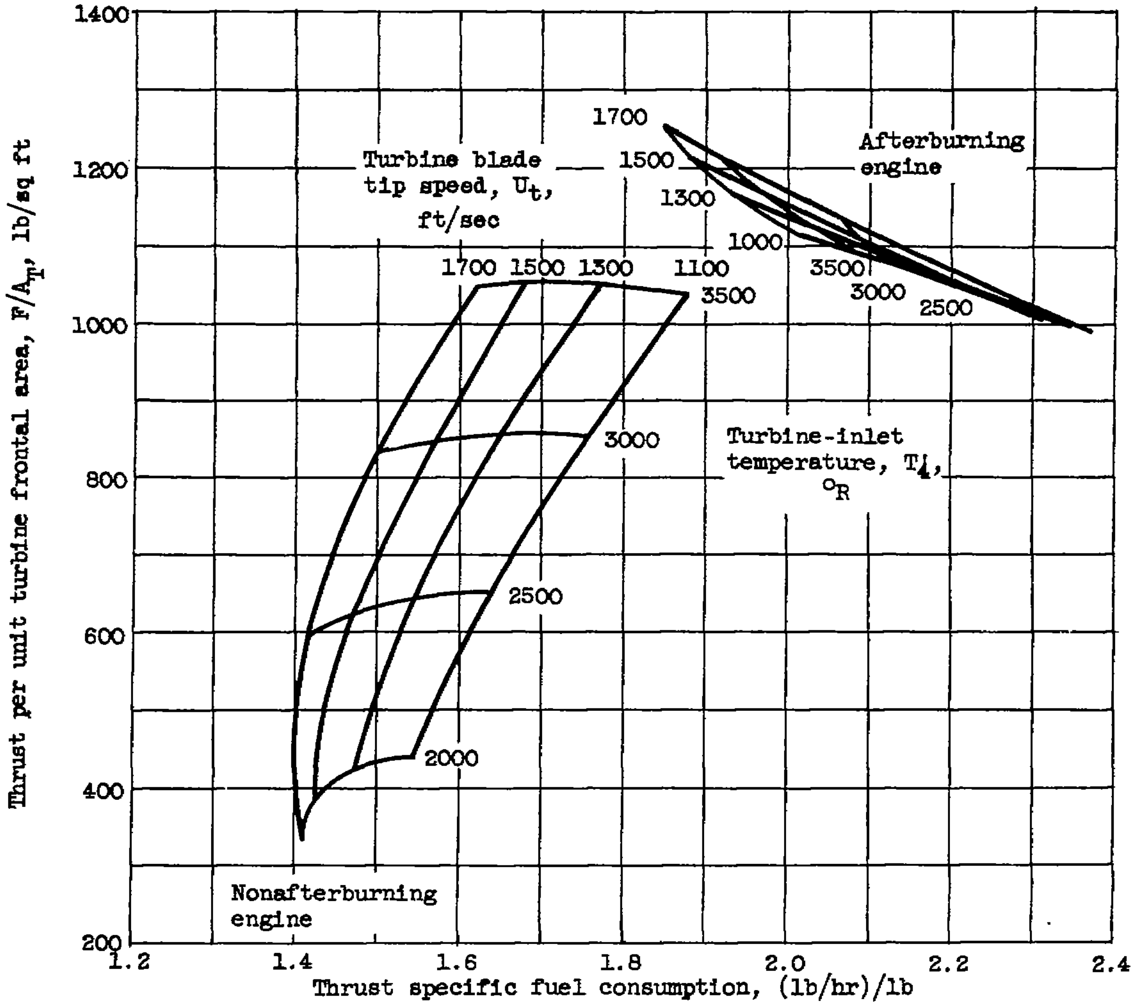

Figure 6. - Afterburning- and nonafterburning-engine performance for range of turbine design variables with constant turbine specific weight flow. Turbine hub-tip radius ratio r_h/r_t, 0.70; turbine-limited specific weight flow $\frac{w_c\sqrt{\theta_2}}{A_T\delta_2}$, 18.5 (lb/sec)/sq ft; afterburner-outlet temperature T_{10}', 3500° R; coolant-flow ratio C, 0.

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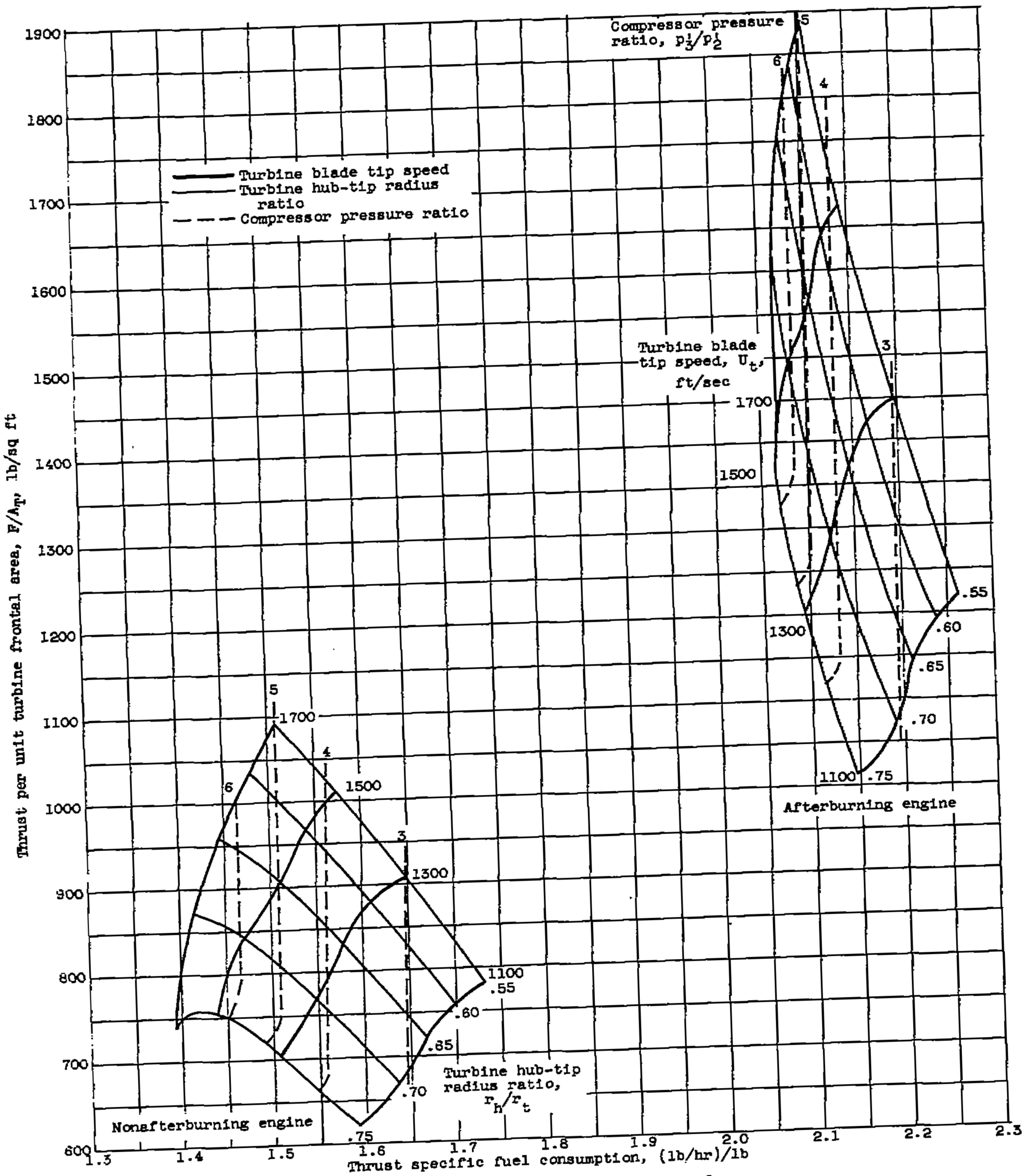

(a) Turbine-inlet temperature, T_4', 2500° R.

Figure 7. - Afterburning- and nonafterburning-engine performance with turbine hub-tip radius ratio. Afterburner-outlet temperature T_{10}', 3500° R; coolant-flow ratio C, 0.

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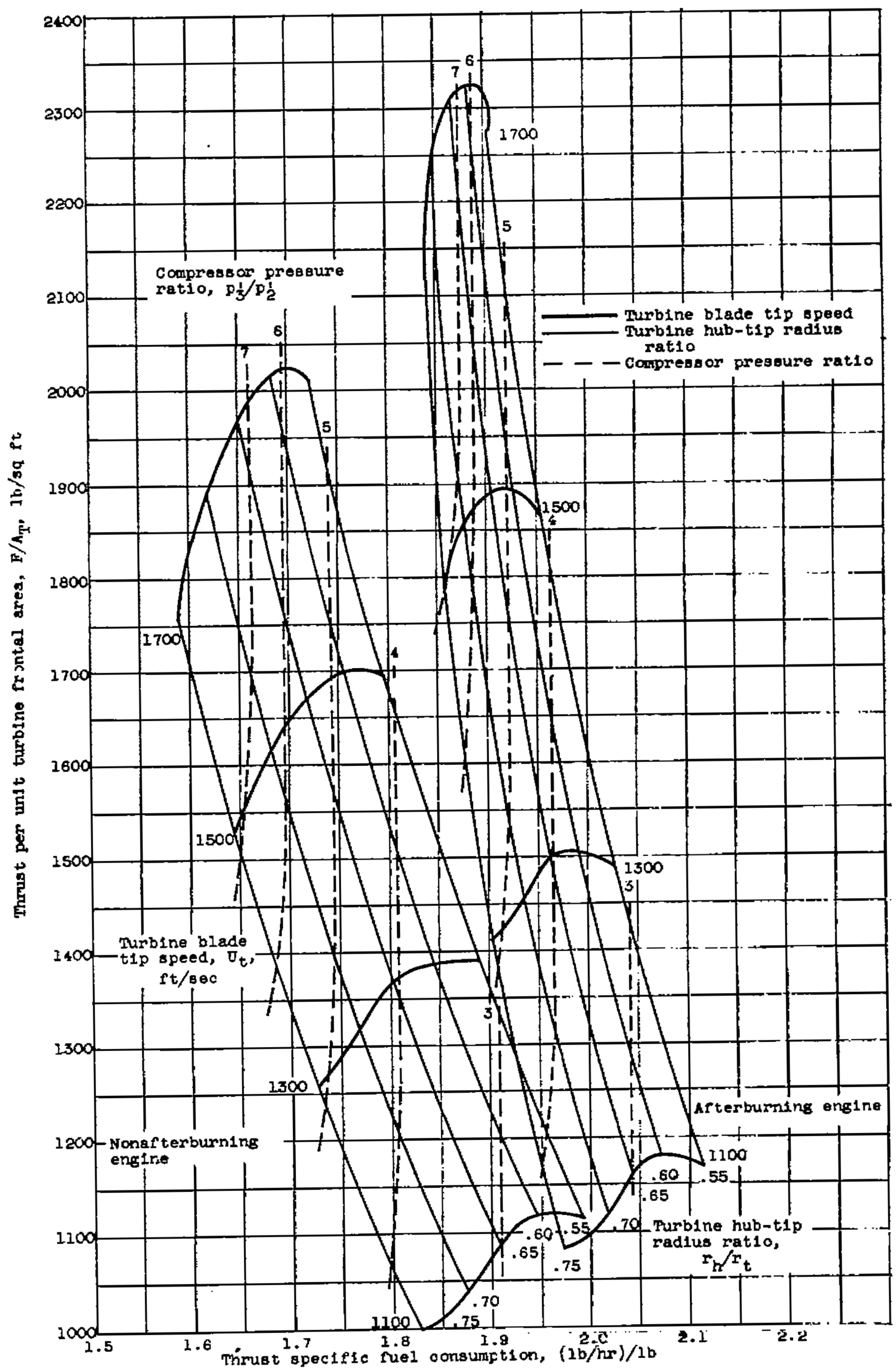

(b) Turbine-inlet temperature, T_4', 3500° R.

Figure 7. - Concluded. Afterburning- and nonafterburning-engine performance with turbine hub-tip radius ratio. Afterburner-outlet temperature T_{10}', 3500° R; coolant-flow ratio C, 0.

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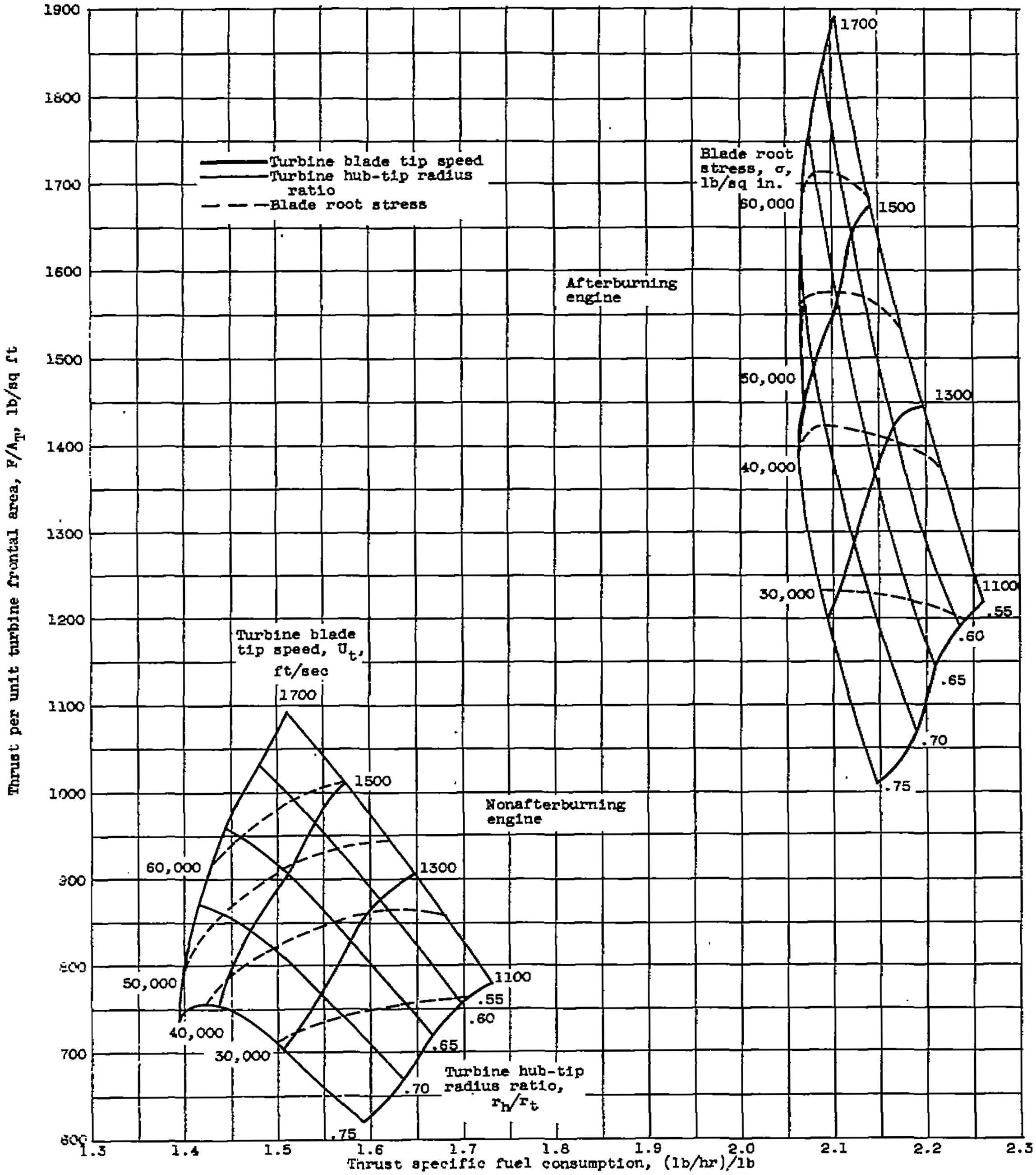

(a) Turbine-inlet temperature T_4', 2500° R.

Figure 8. - Comparison of stresses at various performance levels for afterburning and nonafterburning engines. Afterburner-outlet temperature T_{10}', 3500° R; coolant-flow ratio C, 0.

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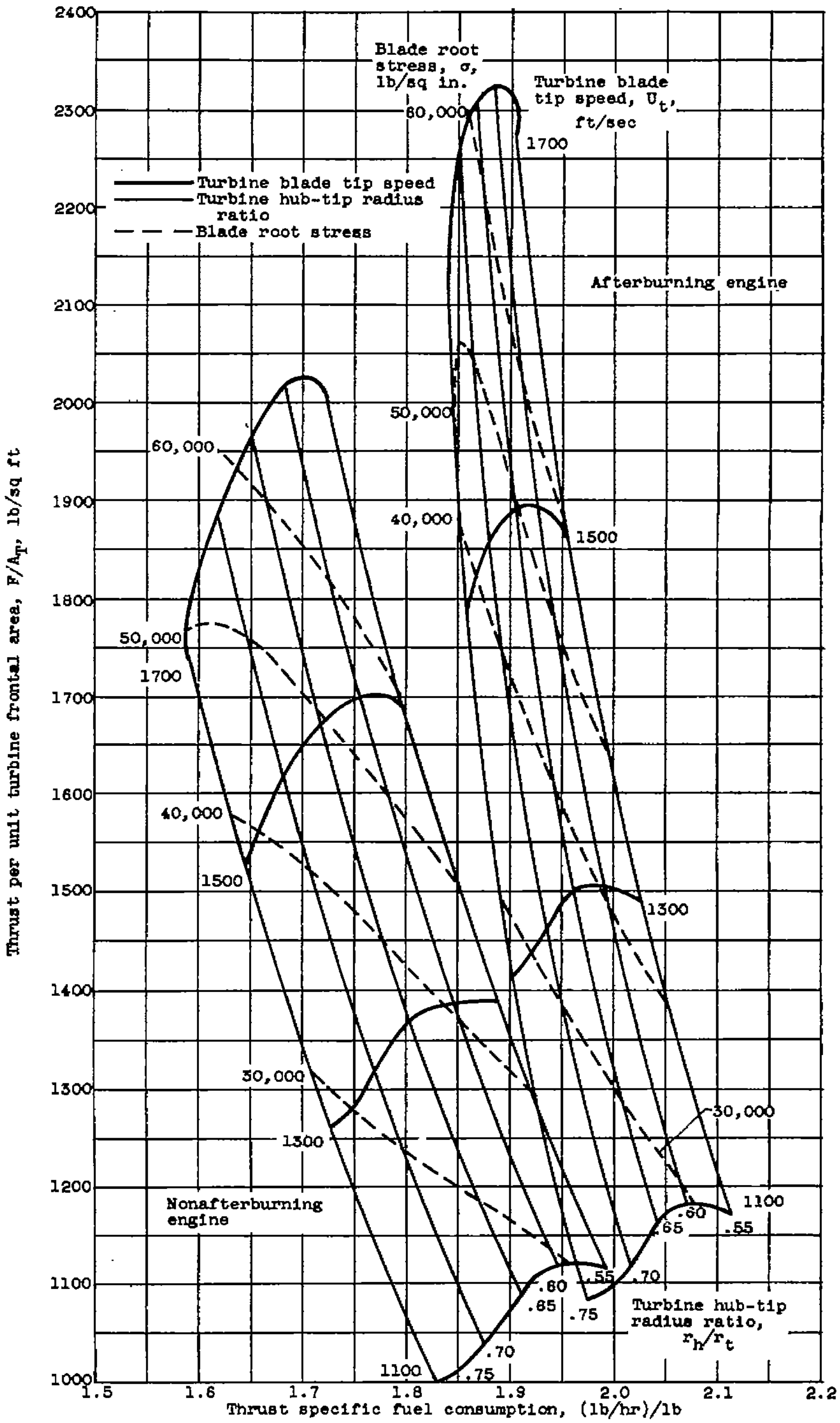

(b) Turbine-inlet temperature T_4', 3500° R.

Figure 8. - Concluded. Comparison of stresses at various performance levels for afterburning and nonafterburning engines. Afterburner-outlet temperature T_{10}', 3500° R; coolant-flow ratio C, 0.

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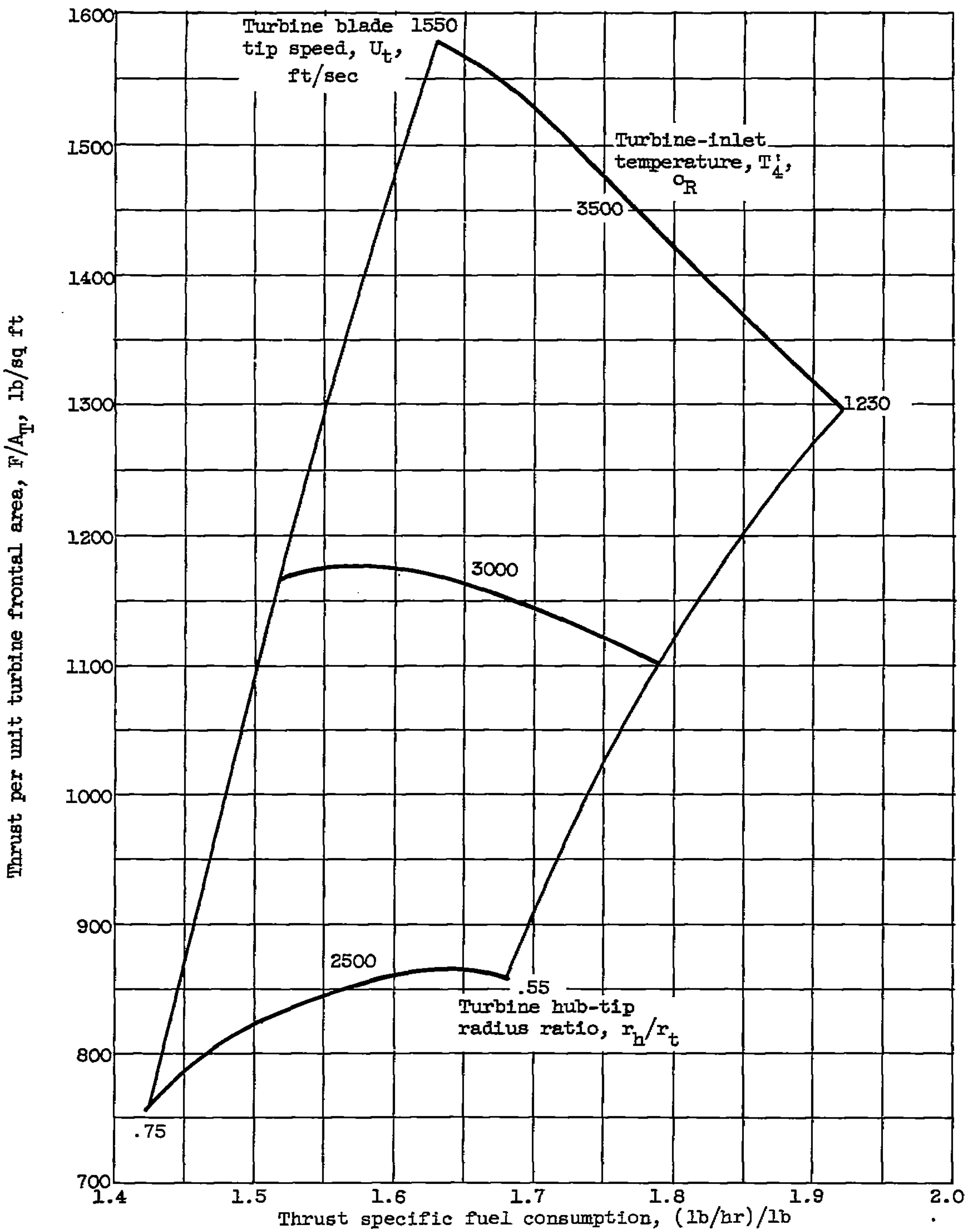

Figure 9. - Variations in nonafterburning-engine performance. Tapered blade-root stress, 40,000 pounds per square inch; coolant-flow ratio C, 0.

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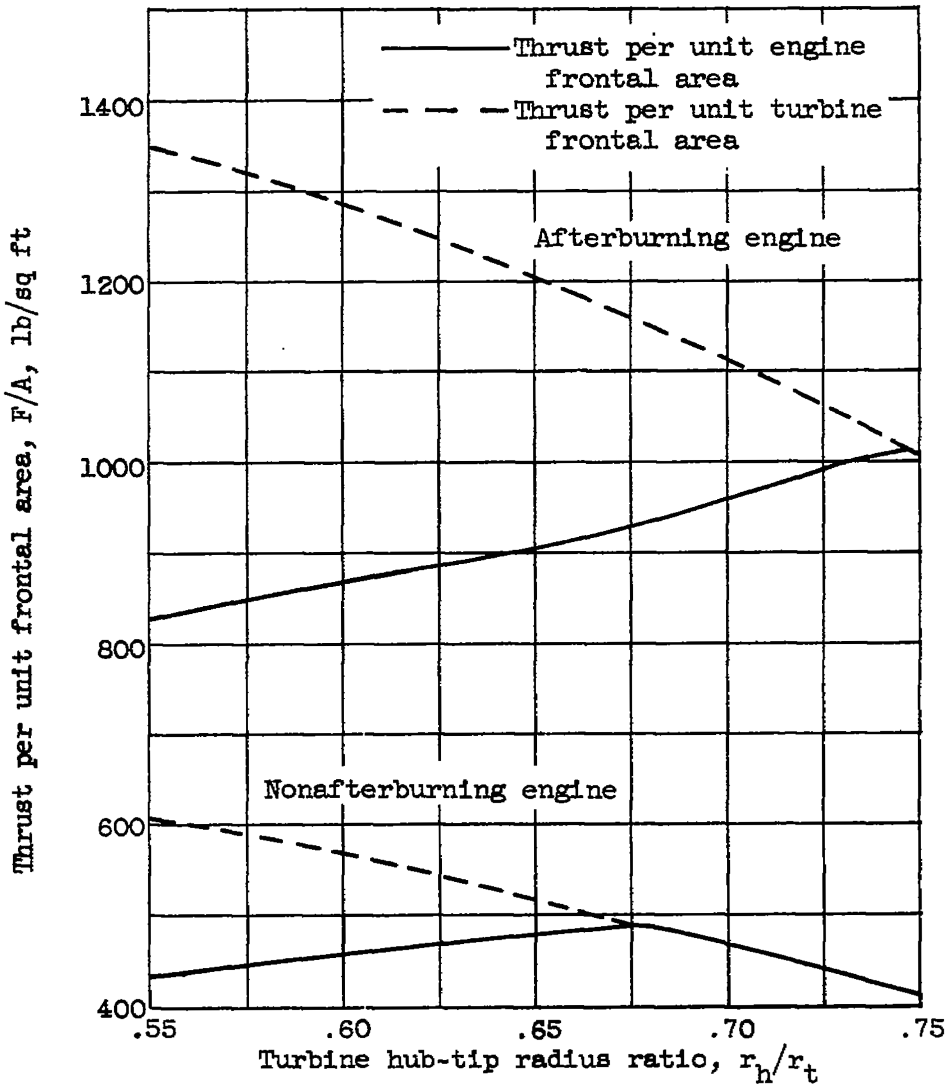

Figure 10. - Limitations on afterburning- and nonafterburning-engine thrusts due to relative engine-component frontal areas. Turbine tip speed U_t, 1300 feet per second; turbine-inlet temperature T_4, 2000° R; coolant-flow ratio C, 0.

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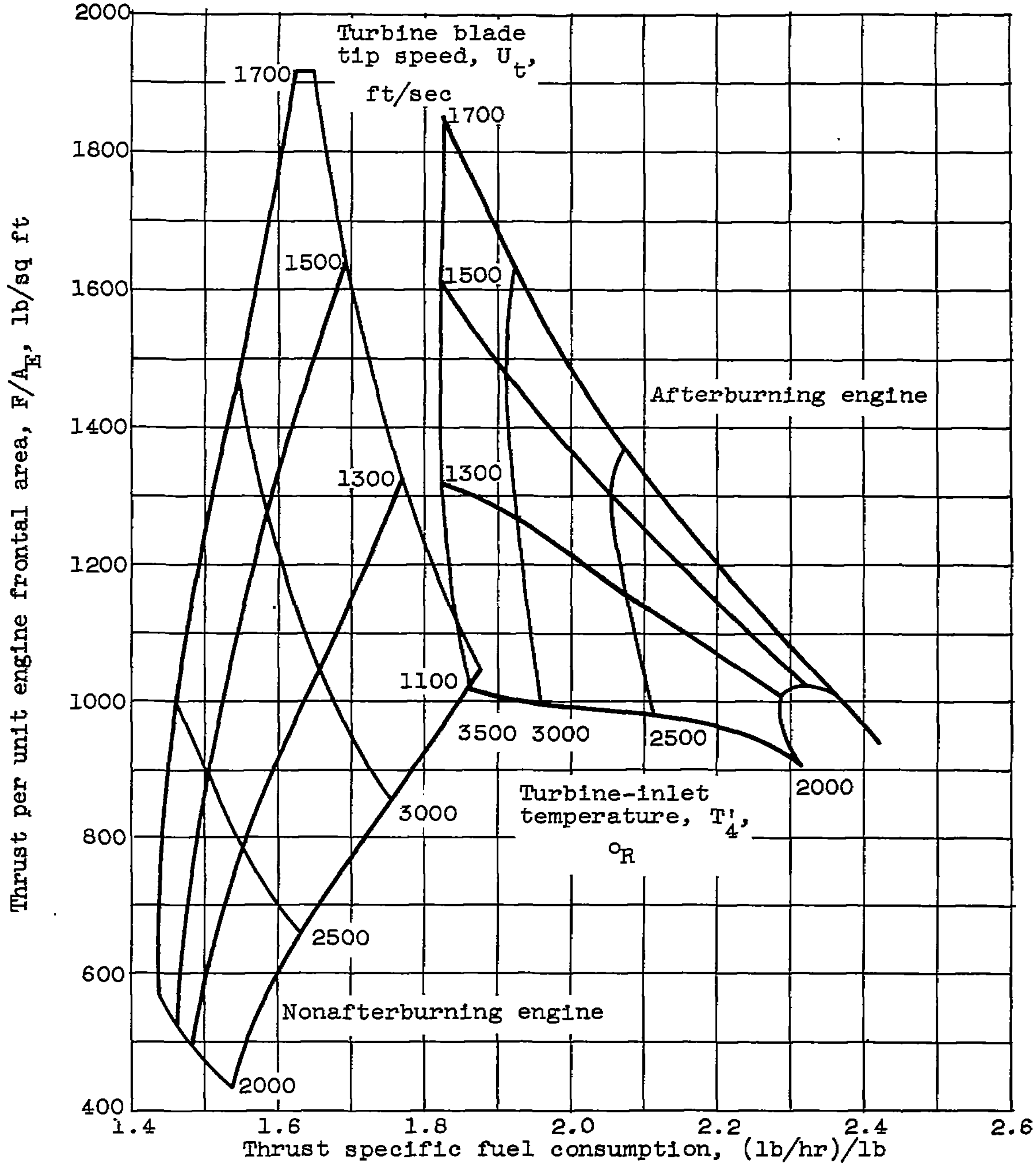

Figure 11. - Maximum afterburning- and nonafterburning-engine thrust per unit engine frontal area with turbine tip speed and inlet temperature. Turbine hub-tip radius ratio varies to correspond to maximum thrust; afterburner-outlet temperature T_{10}', 3500° R; coolant-flow ratio C, 0.

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